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IMPROVED SCALING LAWS FOR STAGE INERT MASS OF SPACE PROPULSION SYSTEMS

Volume I - Summary

June 1971

PREPARED
FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
AMES RESEARCH CENTER
MOFFETT FIELD, CALIFORNIA 94035

CONTRACT NAS2-6045

...DY MANAGER

Space Division
North American Rockwell

SD71-534-1

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APPROVED BY

J. A. BODDY
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FOREWORD

The Improved Scaling Laws for Stage Inert Mass of Space Propulsion Systems Study was conducted by the Space Division of the North American Rockwell under Contract NAS2-6045 for the Advanced Concepts and Mission Division of the National Aeronautics and Space Administration. The contract involved a study for the development of improved scaling laws for stage inert mass of future planetary vehicle systems. The laws were to consider the effects of mission profiles, propulsion/propellant combinations and advanced structural concepts.

This report is submitted in three volumes -

I	(SD71-534-1)	Summary Report
II	(SD71-534-2)	System Modeling and Weight Data
III	(SD71-534-3)	Propulsion Synthesis Program - Users and Programmers Manual

This volume summarizes the improved scaling laws and the simplified scaling laws (manual synthesis) developed in Volume II, and discusses the various subsystem models used for the vehicle synthesis.

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The following North American Rockwell, Space Division, personnel were the major contributors to the technical contents of this volume.

C. M. McCrary	Propulsion and Fluid Systems
C. K. McBaine	Weight Prediction
B. J. Hamry	Structural Synthesis
A. J. Richardson	Penetration Mechanics
J. P. Sanders	Meteoroid Structures
J. A. Boddy	Thermal Insulation
C. W. Martindale	Synthesis Program Development
J. A. Boddy	System Modeling

The Contracting Officer Representative for the National Aeronautics and Space Administration, Duane W. Dugan of the Advanced Concepts and Mission Division, provided valuable guidance and direction throughout the study.

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ILLUSTRATIONS

TABLES

NOMENCLATURE

A	Exposed surface area, in^2
A_A	Effective absorbing area, in^2
A_E	Effective emitting area, in^2
a	Semi-major axis of mission trajectory, AU
B	Albedo of the planet
b/a	Bulkhead aspect ratio
D	Stage diameter, in
d_p	Meteoroid particle diameter, cm
E	Young's modulus of construction material
E^b	Performance mass ratio = $\prod_{k=1}^{b} EXP\ (V_k/Ig)$
$EXP(x)$	Exponential function of x
e	Eccentricity of mission segment trajectory
F	Thrust level, lbf
f	Additional area factor for propellant boil-off = $4/\left(D\rho_i\right)$
G	Additional tank weight for propellant boil-off = $f\ \rho_{ins}\ t_{cyl}$
h	Spacecraft orbit altitude around planet, km
I	Specific impulse of main propulsion, sec
K_i	Total normalized heat absorbed by the i^{th} stage, $Q_i\ d_i/A_i$
K_{21}	Normalized heat absorbed by the second stage between the first and second burn of the vehicle
L	Latent heat of vaporization
L/D	Stage fineness ratio
M	Molecular weight of pressurant

NOMENCLATURE (CONT'D)

MR	Propellant mixture ratio, o/f
N_E	Number of engines per stage
N_x	Compressive load intensity, lb/in
P, Pt	Tank design pressure, lb/in^2
P_c	Engine chamber pressure lb/in^2
P_o	Probability of no penetration
R	Stage or tank radius, in
R_P	Solar distance of planet, AU
r_p	Planet's radius
$S_\oplus$	Solar constant
T_c	Temperature on inner surface of propellant tank, $^\circ$R
T_H	Surface equilibrium temperature, $^\circ$R
T_u	Ullage gas temperature, $^\circ$R
T/W	Ratio of thrust to weight
t_b	Engine burn time, sec
t_{cyl}	Skin thickness of stage shell
UF	Ullage fraction
V	Stage velocity increment, ft/sec
V_L	Volume of liquid, in^3
V_P/V_e	Ratio of velocity of planet to velocity of earth
V_p	Meteoroid particle velocity, km/sec
V_t	Volume of propellant tank, in^3
W_{B_i}	Weight of propellant boil-off for ith stage
W_{JET}	Jettisoned weight, lb
W_{PAY}	Payload weight, lb

NOMENCLATURE (CONT'D)

W_S	Unit weight of structure required for structural integrity, lb/ft^2
$\alpha_{S/\varepsilon}$	Surface coating ratio of absorptivity to emissivity
ΔE	Change in eccentric anomaly
Δt	Mission segment trip time, hr
$\Delta \nu$	Change in true anomaly
ε	Engine expansion ratio
η	Material plasticity correction factor
μ	Performance mass ratio
ρ	Material density lb/in^3
ρ_i ρ^*	Propellant bulk density lb/in^3
σ	Stefan-Boltzmann constant
σ	Material stress level, lb/in^2
τ_{ins}	Insulation thickness, cm
$\dot{\omega}$	Propellant flow rate, lb/sec

SUBSCRIPTS

f	Fuel
o	Oxidizer

1.0 INTRODUCTION

Effective performance evaluation of future space propulsion systems depends on adequate design information describing the weight and performance of the major subsystems of the vehicle. The stage inert mass exerts a significant effect on the attainable level of a system's cost effectiveness and is one of the major design parameters for the determination of the total size of any vehicle system. Weight scaling laws for the subsystem elements must be sufficiently comprehensive to differentiate between types of engines and propellants, types of subsystems and their usage, stage geometric characteristics, special design constraints, and overall mission performance requirements. Scaling laws for structural components must reflect the time period in which the components are to be designed, developed, and utilized.

Scaling laws currently used in weight prediction are based upon specific basepoint designs developed to reflect existing technology. The application of these laws to advanced systems is not always meaningful. Comparative decisions can be based on the relative weight of various concepts, but absolute weight in a usable form is required to identify overall system performance. Reasonably accurate laws are necessary to generate confidence in the performance evaluation of advanced space systems. These scaling laws must be more sophisticated than simple gross laws and must reflect the effects of the controlling design parameters; account for future trends and advances in technology; and be capable of differentiating between materials, types of construction, and system arrangements. An error of a few percent in the stage mass fraction can make the difference between a merely feasible concept and a more efficient concept. Any attempt at weight estimation in the early design phases should provide accuracy, flexibility, and technical depth in sufficient detail to measure the sensitivity of an individual design parameter to each subsystem and to the overall system.

The technology of weight prediction has not kept pace with advances in dynamic analysis, structural analysis, and trajectory performance. Many weight prediction tools are used to extrapolate beyond the allowable region of the basepoint design. Items such as minimum feasible weight due to design constraints, manufacturing constraints, etc., are ignored by the prediction laws and models. The intent of this study was to overcome some of these problems by providing weight modeling of the various systems and subsystem elements of space propulsion modules and to identify the major design parameters that influence scaling laws.

This report summarizes a study, performed by the Space Division of the North American Rockwell Company, which satisfies the need for improved scaling laws for stage inert mass of space propulsion systems. The resulting laws are applicable to current and future vehicle systems and designs for a comprehensive spectrum of anticipated planetary missions.

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2.0 STUDY OBJECTIVES AND APPROACH

Objectives of the study were as follows:

1) Develop improved scaling laws for the stage inerts of cryogenic, space-storable and Earth-storable space propulsion systems over a range of propellant capacities.

2) Generate a set of simplified laws for manual synthesis of space propulsion systems.

3) Provide a computer program for the automated weight synthesis of space propulsion system using these improved scaling laws.

Scaling laws developed for the different subsystems can be used for innumerable vehicle/system combinations of the following types:

1) Design Concepts

   a) Single or Multi-stage; b) Tankage Arrangements

2) Engine

   a) Pressure or Pump-fed; b) Regenerative or Ablative Cooling

3) Propellant Combinations

   a) Cryogenics; b) Space or Earth Storables

4) Material Technology

   a) Conventional; b) Advanced

Propulsion systems were subdivided into independent but interrelated modules. Four major modules were used and weight scaling was developed for their subsystems and primary weight elements as illustrated in Table 1.

Table 1. Major System Modules and Primary Weight Elements

Engine Module	Propellant Module	Environmental Protection	Other Subsystems
Thrust Chamber Assembly	Tank Wall	Cryogen Insulation	Guidance
Thrust Vector Controls	Bulkheads	Insulation Attachment	Navigation
Skirt Enclosing Engines	Forward Skirt	Meteoroid Bumpers	Attitude Control
Thrust-Structure	Intertank	Secondary Structure	Docking
	Aft Skirt		Electrical
	Pressurization		Instrumentation
	Propellant Feed		

Modules were described with respect to the following.

1) Mission objectives
2) Design concept
3) Subsystem selection
4) Design criteria and environment
5) Technology base

Mathematical models were developed to describe the geometric and material properties of the weight elements. Several NR/SD computer programs from the Computer Aided Design (CAD) program library were used to evaluate these elements for families of module size, types of mission, advances in materials and design, and propellant combinations. The mathematical models were compared to known existing systems to provide weight reference points to obtain appropriate nonoptimum weight allowance factors.

A schematic of the overall study approach is shown in Figure 1 and involved the following five basic steps.

1) Data collection
2) Propulsion modeling
3) Structural and environmental synthesis
4) Data reduction for scaling laws
5) Vehicle synthesis procedure to use the improved scaling laws

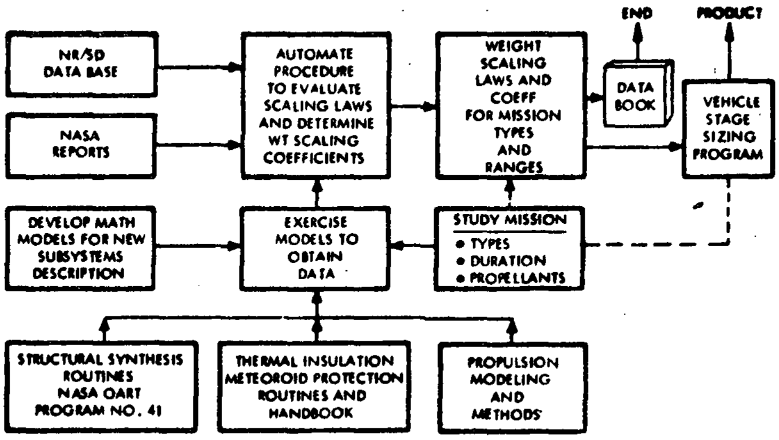


Figure 1. Study Approach Schematic

The study to derive scaling laws for the inert masses utilized the NR/SD weight and design data bank, basepoint designs of vehicles and systems, NASA reports, and results of many in-house advanced technology studies covering various system and subsystem areas. The first study task involved collecting and reviewing all available data to determine where additional parametric data were required. Data were supplemented using mathematical models from several synthesis programs to obtain information relating to advanced designs. Propulsion elements were modeled to identify design parameters of the following four types:

1) Load (gross weight, major externally applied acceleration loads, fuel pressures).

2) Geometry (external-exposed area, structural members, tankage arrangement, thickness).

3) Configuration (type of construction, material, structural arrangement, cutouts).

4) Environment (temperature, external pressure, meteoroid flux, etc.).

Parametric weight data were reduced to a series of scaling laws which describe the system weights as a function of the major design parameters. Data reduction was automatically achieved by a constrained multi-linear regression analysis. The improved scaling laws were then combined in the "Space Propulsion Automated Synthesis Modeling" (SPASM) computer program which provides an effective tool for studying mission requirement effects upon vehicle system weights. A summary of the improved scaling laws is provided in Section 3.0. The simplified scaling laws, Section 4.0, were obtained from design data generated by the SPASM program and condensed from the improved scaling laws.

3.0 IMPROVED SCALING LAWS

3.1 Module Models

The space propulsion systems were divided into four separate modules, each comprised of several elements (structural members or subsystems). Individual mathematical models were developed for the major components to determine the weight sensitivity to important design and mission parameters.

The model for the engine module is a series of multi-parameter scaling laws which have been derived from parametric engine and engine components design data. The design parameters included in the model are the engine class, propellant combination, thrust level, chamber pressure and nozzle expansion ratio. The last three parameters are used to determine the engine diameter and length for the packaging evaluation within the engine module. Scaling laws for the engine thrust chamber assembly are defined for pressure and pump-fed engines using cryogens, space-storable and earth-storable propellants. The outer skirt enclosing the engine(s) is evaluated with respect to its geometry, material and design load environment.

Models for the propellant module include six different arrangements of the main propellant tanks. These include tandem tanks with common or separate bulkheads, spherical or cylindrical tanks with ellipsoidal bulkheads and nested tanks. Weight elements for the propellant module are the tank wall, bulkheads and baffle devices, support structure and the unpressurized skirts, intertank and interstage structure. The necessary design loading environment considers the boost phase of the stage either during Earth ascent fully loaded or carried in the cargo bay of the Earth Orbital Shuttle, and also during space flight under maximum acceleration. Engine propellant requirements will dictate internal pressures for the tanks. Pressurization system models for start-up and steady state requirements include weight estimates for the pressurant, its container if required, and the pressurant transmission.

NR/SD structural synthesis programs were used to identify the unit weight requirements for minimum weight designs for a family of stage diameters and loading intensities with different types of construction and material. This structural synthesis analyzes the designs for their strength, stability (local and general) requirements and considers restrictions due to minimum gages and manufacturing limitation. A multi-linear regression analysis was performed on the structural weight data to provide the scaling laws which are a function of:

1) Loading - Pressure, Axial and Bending Moments

2) Material Properties - Ultimate Stress, Elastic Modulus and Density

3) Construction - Ring Stiffened and Waffle Design

4) Design Criteria - Margin of Safety, Safety Factors (Limit, Ultimate and Burst)

Scaling coefficients are supplied for current materials and types of construction. Advances in structural technology can be easily considered by slightly modifying the scaling coefficients.

The environmental protection module is concerned with the meteoroid and thermal environments and their effect on inert weight and performance degradation with propellant evaporation. Both these fluxes are spatial and time varying throughout the mission and their respective models considered this phenomena by determining the vehicle's distance/time relationship and integrating the fluxes throughout the mission duration.

The meteoroid model contains three separate portions -

1) Meteoroid Flux Integration

2) Meteoroid Particle Determination

3) Shielding Weight Requirements

A NASA environment is used for the flux distribution of sporadic (asteroidal and cometary) and an NR/SD model is used for the stream (cometary) meteoroids. Single sheet penetration mechanics is based on a NASA model while the single- and multi-bumper shielding concepts use an NR/SD empirical penetration model substantiated with extensive test data. When insulation is present, it can also be included in the meteoroid penetration evaluation to help reduce the required shielding weight. Weight data for the outer bumper sheet (if required) and rear shielding sheet were obtained from another NR/SD design optimization program and the parametric weight data are presented in Volume II. Reduction of the data provided the scaling laws with the necessary design parameters and empirical coefficients.

Two different approaches are used for the thermal insulation evaluation:

1) Propellant Boil-off with High Performance Insulation (HPI)

2) Tank Pressure Build-up; No Boil-off with HPI.

Solar flux and planet reflected- and emitted- heat contributions were considered as the prime sources which induced heat into the propellants. A simplified one-dimensional thermal analysis model is employed to estimate the total mission heat input into the propellant tanks and the propellant boil-off.

Thermal protection model considers the absorptivity and emissivity of the outer surface, and the thermal conductivity of the insulation. An effective thermal conductivity is developed from the basic properties of the individual layers of the multi-layer HPI and the temperature differential across the insulation (outer surface temperature and inner tank wall temperature).

Types of insulation supplied with the thermal model are:

1) DAM/NM
2) NRC-2
3) NARSAM
4) Superfloc
5) GAC-9

Propellants considered requiring insulation for some missions are limited to the cryogens and space storables. Such propellants included in this study are:

1) Liquid Hydrogen - LH_2
2) Liquid Oxygen - LO_2
3) Liquid Fluorine - LF_2
4) Oxygen Difluoride - OF_2
5) Diborane - B_2H_6

Insulation, when required, is applied to the tank wall and bulkheads, support structure and the unpressurized shells attached to the tanks. Additional insulation weight allowances are included for the ground hold and earth boost ascent trajectories.

The "Other Subsystem" weight module contains the remaining subsystems and has been modeled using empirical data based upon vehicle size, and mission specifications. Most of these subsystems weight allocations are difficult to define until their actual functional requirements have been identified. The functional activities are strongly dependent on the type of mission and its objectives. Weight from these subsystems will generally contribute only a few percent to the total stage inert mass and errors in their estimation would not have too significant impact on the total stage inert weight and hence performance.

The space mission model is specified in terms of its individual velocity requirements and the trajectory parameters. Each mission is considered as a series of trajectory legs for both the transplanetary and planet stop-over portions of the missions. The trajectory profile is identified by its orbit parameters (semi-major axis, eccentricity or trip duration) and is used to integrate the environmental fluxes (thermal and meteoroid) during the entire mission operation. The velocity requirements are used to determine the propellant loadings for all of the vehicle's stages.

The design models, weight scaling laws and the mission trajectory model are incorporated into the Space Propulsion Automated Synthesis Modeling (SPASM) computer program. This program is capable of exercising all the individual models to synthesize vehicle systems which meet the actual mission and velocity requirements. Program selection and arrangement of the various subsystem models is controlled by the user via an executive main routine.

The SPASM program uses two modes in its synthesis of the multi-stage vehicles, these being:

1) For a given or calculated initial weight (W_O) and mission velocity increment, determine the vehicle's payload capability.

2) For a specified payload and velocity increment, determine the vehicle's gross initial weight (W_O).

The program subroutines are generalized models able to perform the analysis required for most planned planetary missions using current and advanced vehicle systems. Subroutines are provided for the following elements during the synthesis process.

1) Mission integration of meteoroid and thermal flux
2) Propellant requirements based on mission objectives
3) Volume and Surface areas depending on tank arrangement
4) Design environment evaluation, loads, meteoroid and thermal
5) Module weight estimation

3.2 SUMMARY OF IMPROVED SCALING LAWS

Space Propulsion Systems have been categorized into four independent modules for the weight scaling laws. These modules will completely describe the vehicle stage and provide the stage weight and size description. Each of the major modules is composed of several primary subelements as shown in Table 1. The improved scaling laws used by the SPASM program are summarized in this section. The scaling laws are presented here with the English system of units, as used internally by the computer program; the equivalent scaling laws using metric units are provided in Volume II.

3.2.1 Performance Equations

Preliminary sizing is based on an initial estimate for the stage mass fraction ν_B which is used with input data to obtain a starting value for stage initial mass. The stage mass fraction is then varied until the mass of usable propellants calculated from this stage initial mass agrees with that obtained by using the scaling laws to calculate stage inerts and boil-off, or until the variation of ν_B decreases to less than 1×10^{-6}. An updated value of the stage mass is obtained as the sum of the stage inert masses and of the total mass of usable, boil-off and reserve propellants. The procedure is then repeated with the updated value replacing the starting value of the stage initial mass until the difference between the new updated value and the previous value is no greater than 0.01% of the latter, or until the allowable sum of iterations is exceeded.

$$\nu_B = \frac{W_P^*}{W_P^* + W_{ST}}$$

Definition of the terms used in this report may be found in the nomenclature on pages xiii and xiv.

Minimum Initial Weight with Fixed Payload

The propellant requirements of the i^{th} stage for a multi-stage vehicle with multiple burns per stage and propellant boil-off during coast phases of the mission can be expressed as :

$$W_{P_i} = \frac{\left(W_{PAY} + \sum_{k=i+1}^{N} W_{G_k}\right)\left(E^B - 1\right) + \sum_{b=1}^{B-1}\left(E^b - 1\right)\left(W_{JET_{i,b}} + \sum_{k=i+1}^{N}\left(\Delta BO_k\right)_{b+1} W_{P_k}'\right)}{\left[1 - \left(\frac{1}{\nu_B} - 1\right)\left(1 + \Delta BO\right)\left(E^B - 1\right) - \sum_{b=1}^{B-1}\left(\Delta BO_i\right)_{b+i} E^b\right]}$$

where

ΔBO_i is the ratio of propellant boil-off of the i^{th} stage to the usable propellant W_{P_i}

$$\Delta BO_i = \left(\frac{MR_i}{MR_i + 1}\right)\sum_{b=1}^{B}\left(\Delta BO_{ox_i}\right)_b + \left(\frac{1}{MR_i + 1}\right)\sum_{b=1}^{B}\left(\Delta BO_{f_i}\right)_b$$

The total amount of propellant including boil-off propellant for the i^{th} stage is given by

$$W_{P_i}^* = W_{P_i}\left(1 + \Delta BO_i\right)$$

The stage initial weight is

$$W_{G_i}^* = W_{ST_i} + W_{P_i}^*$$

and the total vehicle initial weight prior to any boil-off can be obtained from

$$W_{O_1}^* = W_{PAY} + \sum_{k=1}^{N}\left(W_{ST_k} + W_{JET_k} + W_{P_k}^*\right)$$

Maximum Payload with Fixed Initial Weight

Stage mass sizing for the maximum payload performance for a specified initial launch weight is given by

$$W_P = W_O\left(1 - e^{-V/Ig}\right)$$

and the payload weight

$$W_{PAY} = W_O \quad e^{-V/Ig} - \left(\frac{1}{\nu_B} - 1 \right) W_P$$

If there is propellant boil off from the vehicle stages and the weight prior to ignition and after boil off, W_O, can be estimated to account for the propellant losses, the initial propellant weight is

$$W_P^{*} = W_O \left(1 - e^{-V/Ig}\right) (1 + \Delta BO)$$

and the payload is

$$W_{PAY} = W_O \, e^{-V/Ig} - \left(\frac{1}{\nu_B} - 1 \right) W_P^{*}$$

A difficulty arises in estimating the initial launch weight and an iteration is required to determine the boil-off from as yet an undefined propellant volume.

<u>Stage Inert Weight</u> (W_{ST})

The stage inert weight is made up of the four weight modules and the residual and trapped propellants.

$$W_{ST} = W_{EM} + W_{PM} + W_{EMP} + W_{SYS} + W_{PR}$$

3.2.2 Engine Module

The improved scaling laws for the engine module have been developed for the following system elements.

1) Thrust Chamber Assembly

2) Base Heat Protection

3) Thrust Vector Controls

4) Thrust Structure

5) Skirt Enclosing Engine Cluster

The weight summation of these five elements constitute the engine module weight

$$W_{EM} = W_{ENG} + W_{BHP} + W_{TVC} + W_{TS} + W_{INST}$$

<u>Thrust Chamber Assembly Weight</u> (W_{ENG})

The scaling laws for the thrust chamber assembly have been provided for the various classes of engine system for a series of thrust ranges and propellant combinations

$$W_{ENG} = K_1 F^{K_2} P_C^{K_3} \epsilon^{K_4} t_b^{K_5} + K_6 F^{K_7} P_C^{K_8} \epsilon + K_9 \epsilon + K_{10}: \text{lb}$$

Values for K_1 through K_{10} depend upon the type of engine and thrust level and are shown in Table 2.

Base Heat Protection (W_{BHP})

$$W_{BHP} = K_1 (T/W)^{-0.666} (L/D)^{-0.663} F^{0.7807} ; \text{lb}$$

where $K_1 = 0.0205$ for low radiation propellants $\left(LF_2/LH_2, LO_2/LH_2, LF_2/Li/LH_2\right)$

$K_1 = 0.0277$ for high radiation propellants $(OF_2/B_2H_6, OF_2/CH_4, FLOX/CH_4, N_2O_4/MMH)$

Thrust Vector Control (W_{TVC})

The thrust vector control system weight is directly related to the engine thrust level which it is required to deflect.

$$W_{TVC} = K_1 + K_2 F ; \text{lb}$$

where

$$\left.\begin{aligned} K_1 &= 0 \\ K_2 &= 0.002209 \end{aligned}\right\} \quad \text{Thrust} < 30{,}000 \text{ lbf}$$

$$\left.\begin{aligned} K_1 &= 50 \\ K_2 &= 0.000542 \end{aligned}\right\} \quad \text{Thrust} > 30{,}000 \text{ lbf}$$

Thrust Structure (W_{TS})

$$W_{TS} = 3.6 \times 10^{-3} F (N_E)^{0.3} \quad \text{lb}$$

Interstage Shell (W_{INST})

$$W_{INST} = \pi L_{ENG} D W_{SHELL}$$

Table 2. Engine Weight Scaling Laws (English Units)

$$W_{ENG} = K_1 F^{K_2} P_c^{K_3} \epsilon^{K_4} t_b^{K_5} + K_6 F^{K_7} P_c^{K_8} \epsilon + K_9 \epsilon + K_{10} \text{ (lb)}$$

Propellant Combination	K_1	K_2	K_3	K_4	K_5	K_6	K_7	K_8	K_9	K_{10}	Thrust Range lbf	Remarks
Pressure Fed												
LO_2/LH_2	0.282	0.853	-0.757	0.24	0.297	NA	NA	NA	NA	NA	1000-30000	Ablative Fixed Nozzle
N_2O_4/MMH	2.03	0.538	-0.703	0.206	0.297						1000-30000	Ablative/Radiation Fixed Nozzle
$N_2O_4/UDMH$	0.31	0.853	-0.757	0.24	0.297						1000-30000	Ablative Stowed Nozzle
	2.23	0.538	-0.703	0.206	0.297						1000-30000	Ablative/Radiation Stowed Nozzle
LF_2/LH_2	0.338	0.853	-0.757	0.24	0.297	NA	NA	NA	NA	NA	1000-30000	Ablative Fixed Nozzle
OF_2/CH_4	2.44	0.538	-0.703	0.206	0.297						1000-30000	Ablative/Radiation Fixed Nozzle
OF_2/B_2H_6	0.372	0.853	-0.757	0.24	0.297						1000-30000	Ablative Stowed Nozzle
$FLOX/CH_4$	2.68	0.538	-0.703	0.206	0.297						1000-30000	Ablative/Radiation Stowed Nozzle
Tripropellant	0.008672	1.22	-0.70	0.5	NA	NA	NA	NA	NA	20	1000-30000	
Pump Fed												
LO_2/LH_2	2.5×10^{-5}	1.0	-1.0	2.0	NA	0.0183	1.0	0.0	NA	5.0	1000-8000	Fixed Nozzle
LF_2/LH_2	2.5×10^{-5}	1.0	-1.0	2.0		0.0105	1.0	0.0		80.0	8000-30000	Fixed Nozzle
	2.5×10^{-5}	1.0	-1.0	2.0		0.00966	1.0	0.0		110.0	30000-250000	Fixed Nozzle
	0.015	1.0	-1.0	1.0	0.0	0.0189	1.0	0.0		5	1000-30000	Stowed Nozzle
	0.015	1.0	-1.0	1.0	NA	0.00966	1.0	0.0		110.0	30000-250000	Stowed Nozzle
	2×10^{-8}	1.5	-1.0	1.0		1.02×10^{-5}	1.0	1.0	2.5	1000.0	200000-750000	High Pressure
OF_2/CH_4	1.1819	0.814	-0.43	0.05	NA	NA	NA	NA	NA	NA	1000-50000	Fixed Nozzle
$FLOX/CH_4$	0.41054	0.9269	-0.467	0.094							50000-250000	Fixed Nozzle
	1.30	0.814	-0.43	0.05							1000-50000	Stowed Nozzle
	0.43	0.9269	-0.467	0.094							50000-250000	Stowed Nozzle
Tripropellant	5.94×10^{-3}	1.1	0.0	0.0	NA	1.4×10^{-5}	1.1	0.0	NA	25.0		Fixed Nozzle

NA= Not Applicable

The unpressurized shell weight, W_{SHELL}, is defined in section 3.2.3.

$$L_{ENG} = K_1 + K_2 \left(\frac{F}{P_C}\right)^{1/2} \left(\epsilon^{K_3} - K_4 \right)$$

$$D_{ENG} = K_5 \left(\frac{F}{P_C}\right)^{1/2}$$

Values of the K's for the engine length and diameter are shown in the following table.

Propellant	Nozzle Type	K_1	K_2	K_3	K_4	K_5	Thrust Range
Bipropellant	Fixed	40	1.05	0.5	1.0	0.815	> 10000 lbf
	Fixed	5	1.45	0.5	1.0	0.815	< 10000 lbf
	Stowed	40	0.53	0.5	1.0	0.815	> 10000 lbf
	Stowed	5	0.73	0.5	1.0	0.815	< 10000 lbf
Tripropellant	Fixed	0	2.31	0.4	0.0	0.815	--

3.2.3 Propellant Module

The propellant module consists of the structural elements for the propellant containers, and the support structure between tanks and stages. Weight scaling relationships are quoted for the various types of materials and construction for different environments, pressure and/or compressive stresses, and the component geometry. Six different tank arrangements considered for the synthesis approach are:

1) 2 tandem tanks, identical radii and separate bulkheads
2) 2 tandem tanks, identical radii and common bulkheads
3) Single forward tank and 3 internally suspended aft tanks
4) Single forward tank and 4 internally suspended aft tanks
5) 2 spherical tanks with separate bulkheads
6) Single cylindrical forward tank and aft toroidal tank

<u>Propellant Module Weight</u> (W_{PM})

The module weight is comprized of the various structural and system elements.

$$W_{PM} = W_{TANK} A_{TANK} F_{NOM} F_{NOC} + W_{BULK} F_{NOM} F_{NOC} + W_{SHELL} A_{SHELL} F_{NOM} F_{NOC} + W_{SB} + W_{INTER} + W_{PF} + W_f + W_{PT} + W_{PP}$$

where F_{NOM}, F_{NOC} are tabulated below and A is the surface area of the component

	F_{NOM}			
Construction Material	Monocoque	Skin Stringer	Waffle	Sandwich
Aluminum	1.04	1.05	1.05	1.07
Titanium	1.05	1.06	1.06	1.08
Berylium	1.05	1.06	1.06	1.08
Composites	1.06	1.07	1.07	1.09

	$\bar{F}_{NOC}$
Forward Skirt	1.02
Forward Bulkhead	1.02
Aft Bulkhead	1.05
Tank Wall	1.03
Intertank	1.03
Aft Skirt	1.03
Interstage	1.06

Structural Shell Unit Weights

The design loading intensity, N_x, has to be developed for all the structural components of each model and depends upon the design conditions which are:

1) Earth launch fully loaded as payload of expendable vehicle system
2) Earth launch fully loaded as payload in earth orbital shuttle cargo bay
3) Space launched

The unit structural weights are scaled by

Tank wall

$$W_{Tank} = K_1 \, PR \, \sigma^{-1} + K_2 N_x^{\;K_3} \, R^{K_4} \, P^{K_5} \left(\frac{E}{10^6}\right)^{K_6}$$

Values for K_1 through K_6 are given in Table 3.

Tank bulkhead (total weight)

$$W_{BULK} = K_1 \rho \left(\frac{b}{a}\right)^{K_2} R^3 \quad P \sigma^{-1} \qquad 0.707 < \frac{b}{a} < 1.0$$

$$= K_1 \rho \left(\frac{b}{a}\right)^{K_2} R^3 \quad \left(K_3 - \frac{b}{a}\right)^{K_4} P^{\left(K_5 + \frac{b}{a}\right)^{K_6}} E^{K_7} \qquad 0.5 < \frac{b}{a} < 0.707$$

	K_1	K_2	K_3	K_4	K_5	K_6	K_7
b/a > 0.707	3.14	0.20	-	-	-	-	-
b/a < 0.707	19780	-2.634	0.75	1.049	0.293	0.888	-1.056

Unpressurized shells

$$W_{SHELL} = K_1 N_x^{K_2} \quad \sigma^{K_3} \quad (R+K_4)^{K_5} \quad E^{K_6}$$

Values for K_1 through K_7 are supplied in Table 4

Propellant Baffles (W_{SB})

$$W_{SB} = 0.19 \left(\frac{W_{P_o}}{\rho_o R_o} + \frac{W_{P_f}}{\rho_f R_f} \right)$$

Bulkhead to Tank Wall Attachment (W_{INTER})

Additional weight is considered for the attachment of the bulkheads, tank walls and the outer unpressurized shell.

$$W_{INTER} = 4 \cdot 9 \times 10^{-5} \; F^{1 \cdot 083} \; P_C^{0 \cdot 5}$$

Propellant Feed System (W_{PF})

The weight for the propellant feed systems are itemized for the oxidizer and fuel systems.

Table 3 . Scaling Laws for Pressurized Shell Weights

Construction	K_1	K_2	K_3	K_4	K_5	K_6
I-Section Stringer Aluminum	10.40	0.332	0.533	0.778	0	-3.00
Hat-Section Stringer Aluminum	10.40	0.350	0.533	0.778	0	-3.00
Waffle Aluminum	10.40	0.0216	0.650	0.778	0	-3.00
I-Section Stringer Titanium	16.44	4.50	0.325	0.915	0	-3.00
Hat-Section Stringer Titanium	16.44	1.72	0.660	0.600	0	-3.00
Waffle Titanium	16.44	.084	1.37	-0.30	0	-3.00
I-Section Stringer Beryllium	6.78	1.98	0	1.0	1.0	-3.00
Hat-Section Stringer Beryllium	6.78	2.10	0	1.0	1.0	-3.00
Waffle Beryllium	6.78	1.82	0	1.0	1.0	-3.00

Table 4 . Scaling Laws for Unpressurized Shell Weights

Construction	K_1	K_2	K_3	K_4	K_5	K_6
I-Section Stringer Aluminum	60.8	0.44	0	50.0	0.335	-0.519
Hat-Section Stringer Aluminum	15.5	0.540	0	50.0	0.430	-0.519
Waffle Aluminum	7.1	0.530	0	50.0	0.591	-0.519
I-Section Stringer Titanium	64.46	0.4031	0	50.0	0.3026	-0.465
Hat-Section Stringer Titanium	55.80	0.410	0	50.0	0.295	-0.465
Waffle Titanium	2.45	0.535	0	50.0	0.712	-0.465
I-Section Stringer Beryllium	0.617	1.00	0	50.0	0.189	-0.500
Hat-Section Stringer Beryllium	0.549	1.00	0	50.0	0.189	-0.500
Waffle Beryllium	0.563	1.00	0	50.0	0.189	-0.500

$$W_{PF_i} = K_1 + K_2 N_E \left(\frac{F}{I_{SP}\rho_i}\right)^{1/2} \left(\frac{W_P}{1000}\right)^{K_3}$$

Thrust lbf	Oxidizer			Fuel		
	K_1	K_2	K_3	K_1	K_2	K_3
> 200,000	600	1.10	0.73	880	1.75	0.68
< 200,000	80	5.30	0.73	18	16.7	0.68

Pressurization System

The pressurization system consists of the pressurant gas, its container (if any), and the pressurant transmission.

Pressurant Weight (W_f)

$$W_f = 0.1\ K \left(\frac{P_t V_t}{T_u}\right) M$$

where K = collapse factor $\begin{Bmatrix} 1.0 \text{ single burn} \\ 1.2 \text{ multiple burn} \end{Bmatrix}$

Pressurant Tank Weight (W_{PT})

If the pressurant used for pressurization is propellant from the main tank which has been cycled through a heat exchanger, then there will be no tank required to contain the pressurant. Otherwise, separate pressurant tanks must be supplied and their weight scaling relationships are

$$W_{PT} = 1.4\,\rho_m \left[\frac{1.5 P \eta W_f \ (1 + 2L/D)}{\rho_f \sigma (1 + 1.5\ L/D)}\right] \quad \text{- liquids storage}$$

$$= 1.4\,\rho_m \left[\frac{18 \eta W_f RT_f \ (1 + 2\ L/D)}{\sigma\ (1 + 1.5\ L/D\ )}\right] \quad \text{- gas storage}$$

Pressurant Transmission Weight (W_{PP})

The plumbing valves, heaters, regulators, etc., which constitute the pressurant transmission can be scaled from

$$W_{PP} = K_1 \exp \left(\frac{K_2 \dot{\omega}}{\rho_i} \right)^{0.125}$$

K_1 = 10·70 pump-fed; 18·06 pressure-fed

K_2 = 1565

3.2.4 Environment Shielding Module

The meteoroid shielding unit weight is to be applied to all of the exposed outer structure (A_{mp}). The tank bulkheads are assumed to be shielded by adjacent surrounding structure.

Insulation is applied to the walls and bulkheads of the tanks which contain propellant requiring thermal protection. The unpressurized shells adjacent to the tank will result in heat leaks into the tank. These heat leaks are minimized by providing structural heat blocks and insulating the outer surfaces of the structure.

$$W_{EMP} = W_{mp} A_{mp} + W_{INS} A_{INS}$$

<u>Meteoroid Shielding Weight</u> (W_{mp})

The meteoroid flux density distribution and modification factors due to the presence of planets are based on the models detailed in the NASA Meteoroid Environment Model - 1970 (Interplanetary and Planetary). The mission profile and duration determine the particle's diameter and velocity which must be resisted by the vehicle's meteoroid shielding.

The shielding weight depends on the particle diameter, density and velocity and the type of shielding concept (single sheet, or multiple bumper). Shielding weight is given as the weight of the rear sheet W_{m2} and weight of the outer bumper, W_B.

$$W_{m2} = K_1 \; d_p^{\alpha} \; V_p^{\beta}$$

$$W_B = \text{Maximum} \left[K_2 \, d_p \; , \; K_3 + \left(\frac{K_4 - d_p}{K_5} \right) \left(\frac{V_p - K_6}{V_p} \right) \right]$$

Values for K_1 through K_6, α and β are supplied for the single sheet (Table 5), single bumper (Table 6) and dual bumper (Table 7).

Table 5. Scaling Coefficients - Single Sheet

α	β	K_1	Material	Meteoroid
1.0535	0.667	0.713	Aluminum	Asteroidal
1.0535	0.667	0.825	Titanium	Asteroidal
1.0535	0.667	0.600	Glass Epoxy	Asteroidal
1.0535	0.667	0.259	Aluminum	Cometary
1.0535	0.667	0.300	Titanium	Cometary
1.0535	0.667	0.218	Glass Epoxy	Cometary

Actual meteoroid shielding unit weight penalty (W_{mp}) considers the shell material, unpressurized/pressurized and insulation shielding allowances

$$W_{mp} = W_B + \frac{K_1 \; K_2 \; W_{m2}}{\text{EXP}\;(14.9 \; \rho_{ins} \; \tau_{ins}/d)} - W_S$$

where K_1 = 1.0 for pressure tanks
= 0.445 for unpressurized shells
K_2 = 1.0 aluminum
= 1.15 titanium
= 0.83 Glass Epoxy

All the design parameters, weights and meteoroid dimensions and velocities are quoted in metric units to agree with the coefficients in Tables 5 through 7. The final unit weight W_{mp} should be converted to lb/ft^2.

Thermal Insulation Weight (W_{INS})

The required insulation thickness is obtained either by optimizing for minimum stage weight or stipulating the percent boil-off or providing sufficient insulation and allowing the tank pressure to increase with no propellant boil-off.

Insulation unit weight W_{INS} is given by

$$W_{INS} = \rho_{INS} \; d + \Delta W_{INS}$$

Table 6 . Scaling Coefficients for Single Bumper

METEOROID	MATERIAL	VELOCITY m/sec	K_1	K_2	K_3	α	β
Cometary	Aluminum	V > 8000	0.0412	1.77	1.71	1.12	0.667
Cometary	Aluminum	V ≤ 8000	2225	1.77	1.71	1.12	-0.546
Cometary	Titanium	V > 8000	0.0332	3.18	2.77	1.11	0.667
Cometary	Titanium	V ≤ 8000	4360	3.18	2.77	1.11	-0.645
Cometary	Gloss Epoxy	V > 8000	0.0467	2.28	1.27	1.09	0.667
Cometary	Gloss Epoxy	V ≤ 8000	1092	2.28	1.27	1.09	-0.453
Asteroidal	Aluminum	V > 8000	0.105	5.5	1.71	1.11	0.667
Asteroidal	Aluminum	V ≤ 8000	8960	5.5	1.71	1.11	-0.6
Asteroidal	Titanium	V > 8000	0.0866	7.7	2.77	1.12	0.667
Asteroidal	Titanium	V ≤ 8000	7270	7.7	2.77	1.12	-0.595
Asteroidal	Gloss Epoxy	V > 8000	0.139	8.01	1.27	1.09	0.667
Asteroidal	Gloss Epoxy	V ≤ 8000	4205	8.01	1.27	1.09	-0.485

Table 7 . Scaling Coefficients for Dual Bumper

METEOROID	MATERIAL	VELOCITY m/sec	K_1	K_2	K_3	K_4	K_5	K_6	α	β
Cometary	Aluminum	V > 8000	0.0073	3.01	2.71	0.9	0.52	15000	1.15	.667
Cometary	Aluminum	V ≤ 8000	7.445×10^9	3.01	2.71	0.9	0.346	3550	1.15	-2.45
Cometary	Titanium	V > 8000	0.0062	2.32	2.32	1.0	.286	7350	1.09	.667
Cometary	Titanium	V ≤ 8000	1.42×10^{10}	2.32	2.32	1.0	1.8	4000	1.09	-2.5
Asteroidal	Aluminum	V > 8000	0.0165	6.87	2.71	0.3	2.0	0	1.04	.667
Asteroidal	Aluminum	V ≤ 8000	3.77×10^{10}	6.87	2.71	9.3	2.01	16000	1.04	-2.5
Asteroidal	Titanium	V > 8000	0.0098	9.9	4.95	0.5	2.0	0	1.12	.667
Asteroidal	Titanium	V ≤ 8000	5.59×10^{10}	9.9	4.95	0.5	4.0	0	1.12	-2.6

Insulation	GAC-9	Superfloc	NarSam	NRC-2	Dam/NM
Installed Density ρ_{INS} lb/ft^3	2.71	1.64	2.22	2.17	2.94

$\Delta W_{ins} = 6 \times 10^{-5}$ kg/cm^2; 0.12 lb/ft^2 for ground hold

$= 2 \times 10^{-4}$ kg/cm^2; 0.40 lb/ft^2 exposed during Earth boost to aerodynamic pressure forces

$= 0.0$ for space exposure with an outer meteoroid bumper

Insulation Thickness and Propellant Boil-off

The insulation thickness and propellant boil-off can be optimized in terms of the overall performance. The optimization of the oxidizer and fuel containers can be treated separately.

A two-stage vehicle will have different insulation thicknesses for each propellant container, depending upon the relative mission performance requirements and coast duration for each stage. Optimum insulation thicknesses for the first and second stage of a two-stage vehicle (single burn) are

$$d_{1_{opt}} = \frac{f_1 K_1}{L_1} + \frac{1}{L_1} \sqrt{\frac{L_1 K_1}{\rho_{ins}} \left(G_1 + \frac{1}{\mu}\right) + f_1^{\,2} K_1^{\,2}}$$

$$d_{2_{opt}} = \frac{f_2 K_2}{L_2} + \frac{1}{L_2} \sqrt{\frac{L_2 K_2}{\rho_{ins}} \left[G_2 + \frac{1}{\mu_1 \mu_2}\left(1 + \frac{K_{21}(\mu_1 - 1)}{K_2}\right)\right] + f_2^{\,2} K_2^{\,2}}$$

Corresponding optimum propellant boil-off weights are

$$W_{B_{1_{opt}}} = A_1 \sqrt{\frac{K_1 \rho_{ins} \mu_1}{L_1 (G_1 \mu_1 + 1) + f_1^{\,2} K_1 \mu_1 \rho_{ins}}}$$

$$W_{B_{2_{opt}}} = A_2 \sqrt{\frac{K_2 \rho_{ins} \mu_1 \mu_2}{\left\{ L_2 \left(G_2 \mu_1 \mu_2 + 1 + \frac{K_{21}}{K_2} (\mu_1 - 1) \right) + f_2^{\,2} K_2 \mu_1 \mu_2 \rho_{ins} \right\}}}$$

The results for a single stage vehicle are identical to the first stage of a two-stage vehicle.

The normalized heat flux K is given by

$$K_i = \frac{Q_{IN_i} d}{A}$$

Heat input should be increased to account for heat leaks through the support structure

Recommended 50% for aluminum
40% for titanium
25% for support structure with heat blocks

Integration of the solar heat flux throughout the mission leg will provide the heat input Q_{IN}

$$Q_{IN} = \frac{A}{d} \frac{8766}{2\pi} \left[\frac{C_1 a\sqrt{1-e^2} (\Delta E) + C_2 (\Delta \nu)}{\sqrt{a(1 - e^2)}} - C_3 (\Delta t) \right] +$$

$$\left(\frac{S_{\oplus} \left(1+B_{Eff}\right)}{\sigma R_P^{\,2}} + E_{ff} \right) t_{Stay} \left(\frac{\alpha_S}{\varepsilon} \right) \left(\frac{A_A}{A_E} \right)$$

$$C_1 = \frac{A^*}{2}\left[\left(\frac{\alpha_s}{\varepsilon}\right)\left(\frac{A_A}{A_E}\right)\left(\frac{S_\oplus}{\sigma}\right)\right]^{0.5}$$

$$C_2 = \frac{B^*}{4}\left[\left(\frac{s}{\varepsilon}\right)\left(\frac{A_A}{A_E}\right)\left(\frac{S_\oplus}{\sigma}\right)\right]$$

$$C_3 = \frac{A^*}{2} T_C^{\,2} + \frac{B}{4} T_C^{\,4}$$

t_{Stay} = stay time at planet (hrs)

The heat flux contribution from the planets albedo is obtained from

$$B_{Eff} = B\left(1 - \frac{\sqrt{h\,(2r_p + h)}}{(r_p + h)}\right)$$

and the planet's emitted radiation contribution to the heat flux is

$$E_{ff} = \frac{6.352 \times 10^{10}}{R_p{}^2}\left(1 - \frac{\sqrt{h\,(2\,r_p + h)}}{(r_p + h)}\right)(1 - B)$$

The thermal conductivity of the multi-layer high performance insulation is represented by

$$k = \frac{1}{T_H - T_C}\int_{T_C}^{T_H} (A^*T + BT^3)\, dT$$

where $A^* = 1.168\text{x}10^{-13}\, K_C\, N^{2.725}$

$B = 8.68\text{x}10^{-12}\, K_r\, N^{-1}$

No Propellant Boil-off with Increased Tank Pressure

For the case of allowing the tank pressure to increase and have no propellant boil-off the heat input per volume of propellant Q/V_L is expressed empirically by

$$Q/V_L = K_1\ \text{EXP}\left(\frac{K_2}{K_3 - P}\right)\ UF^{1.25}\quad ;\quad \frac{BtU}{ft^3}$$

Propellant	K_1				K_2	K_3
	LH_2	LO_2	LF_2	B_2H_6		
	0.305×10^{-4}	2.84×10^{-4}	3.6×10^{-4}	3.06×10^{-4}	13.5	15.0

The insulation thickness (d) required is

$$d = \frac{K}{(Q/V_L)} \frac{A}{V_L}$$

K = normalized heat input for the tank

3.2.5 Other Systems Module

Generally other subsystem weights are included in the stages inert mass estimation. Most of these subsystem weights cannot be rigorously evaluated until their particular operational mode and functional requirements are specified. Empirical weight estimates are based on existing hardware designs and proposed studies of discrete basepoint concepts.

The remainder of the vehicle subsystems can be considered by estimating the weights for the intelligence module (W_{IM}), attitude control system (W_{ACS}), docking mechanisms (W_{DOCK}) if required, electrical power and supply (W_{ELECT}), and structure for attaching modules in parallel (W_{ATTACH}). Therefore subsystems weight (W_{SYS}) is approximated by

$$W_{SYS} = W_{IM} + W_{ACS} + W_{ELEC} + W_{DOCK} + W_{ATTACH}$$

<u>Intelligence module weight</u> (W_{IM})

$$W_{IM} = K_1 + D\,K_3\,(K_2 + D\,K_4)\quad W_P^{*}$$

Intelligence Module	K_1	K_2	K_3	K_4
Multi-purpose Independent of Ground Control	1200	.0083	39.2	1.35×10^{-4}
Single Purpose Ground Base Control	200	.0083	23.2	1.35×10^{-4}

Attitude Control System (W_{ACS})

Attitude control system is composed of the control expellants and the associated hardware.

$$W_{ACS} = 150 + 0.002 W_G$$

$$+ \left[0.037 \left(\frac{\Delta V}{I_{SP}} \right)_{ACS} + 1.5 \times 10^{-4} t_{coarse} + 7.5 \times 10^{-2} \, t_{fine} \right] W_O$$

where t_{coarse} and t_{fine} are the days for coarse and fine control

Docking Mechanism (W_{DOCK})

The docking mechanism is applied only to the upper stage where there is attachment and reattachment of the vehicle to the payload. Two basic designs are considered, the Apollo drogue and probe, and the NASA neuter concept for heavier vehicles.

$$W_{DOCK} = K_1$$

Docking Type	Unpressurized K_1	Pressurized K_1
Drogue and Probe	175	200
Neuter	480	520

Electrical System (W_{ELEC})

Electrical wiring, junction boxes, switches, etc., are a function of the stage length, type and number of measurements, and electrical performance required during the mission.

$$W_{ELEC} = K_1 + K_2 W_P^* - K_3 W_P^{*2}$$

Bulk Density	Thrust Level	K_1	K_2	K_3
LOW	< 30000 lbf	0	0.0147	0
	> 30000 lbf	300	4.78×10^{-3}	2.36×10^{-9}
HIGH	< 30000 lbf	0	0.011	0
	> 30000 lbf	300	9.7×10^{-4}	7×10^{-11}

Parallel Attachment Weight

Propellant modules or stages can be attached in parallel to other modules. The weight of the attachment mechanism and local strengthening of the stage depends upon the module diameter and the loads transmitted from one module to the center core module. Loads in the attachment structure have to be transformed from a concentrated load into the module structural shell. The weight increment to each module for the parallel staging is

Center Core Module

$$W_{T_{ATTACH}} = K_2 K_1 F^* D \, SF \quad ; \text{(lb)}$$

Each Outer Module

$$W_{T_{ATTACH}} = 1.25 K_2 F^* D \, SF$$

where K_1 = No. of outer modules attached to center core module

K_2 = 1.59×10^6

F^* = Maximum $\left\{ W_{O_{Module}} \left(T/W_O\right)_{Max} , F_{Module} \right\}$

D = Diameter of module ; in

SF = Safety factor

4.0 SIMPLIFIED SCALING LAWS

The improved scaling laws developed during this study were intended to be used by the SPASM synthesis program. A subset of these laws in a greatly simplified form are provided in this section. The simplified scaling laws are ameniable to manual manipulation for the synthesis of individual vehicle stages while still considering the major design and mission parameters.

The procedure described below is recommended to obtain the performance of a single stage or of a multi-stage system. One stage at a time is considered in systems having more than one stage. If the mission payload is known, synthesis starts with the last stage to be used. If the initial gross mass is known, the first stage to be ignited is synthesized first to obtain the initial mass of the next stage and so on until the payload of the last stage is determined. The synthesis procedure involves iteration through the following five steps:

1. Calculation of total propellant weight.
2. Propellant module inert weight evaluation, W_{PM}
3. Engine module weight estimation, W_{EM}
4. Environmental module weight calculation, W_{EMP}
5. Other system module weight estimation, W_{SYS}.

An initial estimate is made for the mass fraction, $\nu_B = W_P^* / (W_P^* + W_{ST})$, where W_P^* is the sum of the weights of usable and boil-off propellants.

The weights of the four modules calculated later in steps 2 through 5, and the residual propellant weight, W_{PR}, are used to obtain the stage inert mass, W_{ST},

where

$$W_{ST} = W_{PM} + W_{EM} + W_{EMP} + W_{SYS} + W_{PR}$$

The stage mass fraction is computed next and compared with the initial estimate of the stage mass fraction. If the estimated and calculated values of the mass fraction are not within a specified tolerance (0.001), then steps 1 through 5 are repeated with an updated estimate of the mass fraction until convergence is obtained. When one stage has been successfully evaluated the complete process is repeated for subsequent stages.

During the iteration loop for convergence of the stage mass fractions, module weights are expressed in terms of the propellant loading of the stage. This procedure greatly reduces the amount of calculation and table look-up required during the iteration loop. A final check of the module scaling law can be conducted during the stage iteration if the mass fraction greatly departs from the initial mass fraction estimate.

4.1 Total Propellant Weight

The propellant weight requirement (W_P*) is obtained from specifying the fraction of usable fuel that will boil off, ΔBO_f, and the fraction of usable oxidizer that will boil off, ΔBO_{ox}, and using the following equations.

$$\Delta BO = \left(\frac{MR}{MR+1}\right) \Delta BO_{ox} + \left(\frac{1}{MR+1}\right) \Delta BO_f$$

Where ΔBO is the ratio of the total weight of boil-off to the weight of usable propellants. The term MR is the mixture ratio by weight, oxidizer/fuel. The initial weight of propellants W_P*, including that which will boil-off, is given by

$$W_P^* = W_P \, (1 + \Delta BO)$$

Where W_P, the usable propellant, is calculated from one or the other of the two following equations, depending upon whether the mission payload W_{PAY}, or the initial gross mass W_O (including payload and any other upper stages) is given.

For given payload

$$W_P = \frac{W_{PAY} \left(e^{V/Ig} - 1 \right)}{1 - \left(\frac{1}{\nu_B} - 1\right) (1 + \Delta BO) \left(e^{V/Ig} - 1\right)}$$

For given initial gross mass

$$W_P = W_O \left(1 - e^{-V/Ig}\right)$$

Where I is the specific impulse of the stage, Ig is the jet velocity c of the rocket-engine exhaust, and V is the total velocity increment to be supplied by the stage.

Residual and reserve propellants are considered as part of the inert total weight W_{ST}, since they receive the same velocity increment as the tank and other structures generally thought of as inert weight. For obtaining the weight of the propellant module, expressed as a function of its total propellant capacity, these propellants as well as an ullage volume should be taken into account as shown in the next step.

4.2 Propellant Module Inert Weight

The inert weight of the propellant module W_{PM} consists of the weights of propellant tanks, W_{PT}, of the pressurization systems, W_{PRESS}, and of the propellant feed systems, W_{PF}, all for both oxidizer and fuel where applicable.

$$W_{PM} = W_{PT} + W_{PRESS} + W_{PF}$$

Propellant Module Structural Weight (W_{PT})

There are many different design concepts, materials and construction, loading conditions and pressure ranges which can influence the structural weight estimate for the propellant module. For the simplified laws a conventional basepoint design has been considered.

Figure 2 provides the weight estimated for the propellant module structure W_{PT}, for either LO_2/LH_2 or LF_2/LH_2 propellant and indicates the weight variations with mixture ratio for both the separate and common bulkhead design tank arrangements. The effects of changing to the denser fuel combinations are shown in Figure 3 which has weight data for stages using space-and earth-storable propellant combinations. The latter propellants are assumed for a pressure-fed engine system which have tank pressures ranging from 7.03 kg/cm^2 (100 lb/in^2) to 21.09 kg/cm^2 (300 lb/in^2). These high pressures would produce prohibitively heavy designs for the larger stage diameters; therefore the pressure-fed engine systems should be limited to stages with initial gross weight less than 45,000 kg (100,000 lb).

To obtain W_{PT} from Figure 2 , the total propellant capacity of both tanks $W_{P_{tot}}$ should be used. This capacity is computed from the following equations

$$W_{P_{tot}} = W_{P_{tot,ox}} + W_{P_{tot,f}}$$

For oxidizer tank:

$$W_{P_{tot,ox}} = W_P \left(\frac{MR}{MR+1}\right) \left[1 + \Delta BO_{ox} + W_{PR_{ox}} + W_{PRV_{ox}}\right] x \left[1 + UF_{ox}\right]$$

For fuel tank:

$$W_{P_{tot,f}} = W_P \left(\frac{1}{MR+1}\right) \left[1 + \Delta BO_f + W_{PR_f} + W_{PRV_f}\right] \left[1 + UF_f\right]$$

Where, in addition to the terms already defined in Section 4.1,

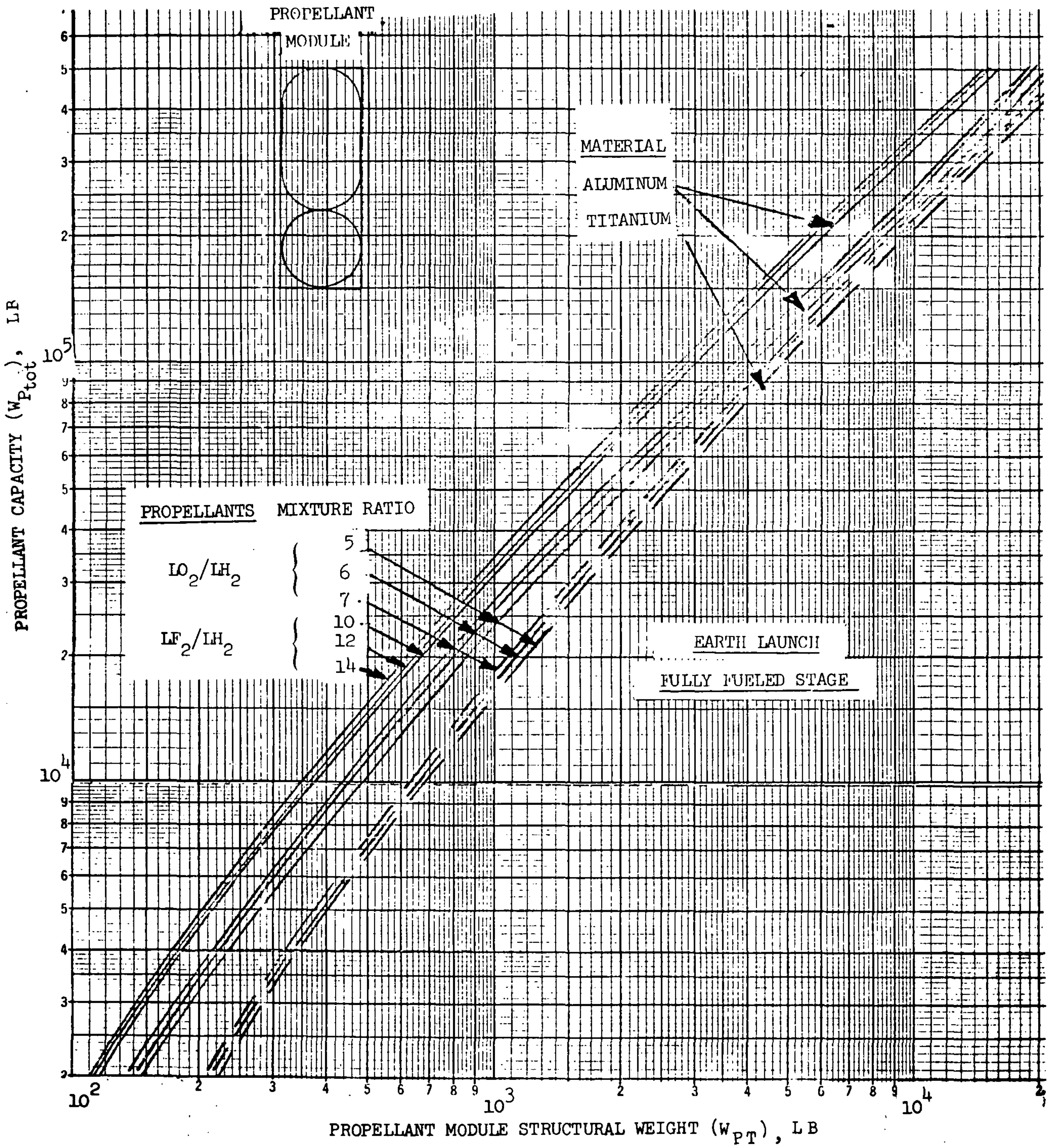


Figure 2. Propellant Module Structural Weight - Cryogen Fuels

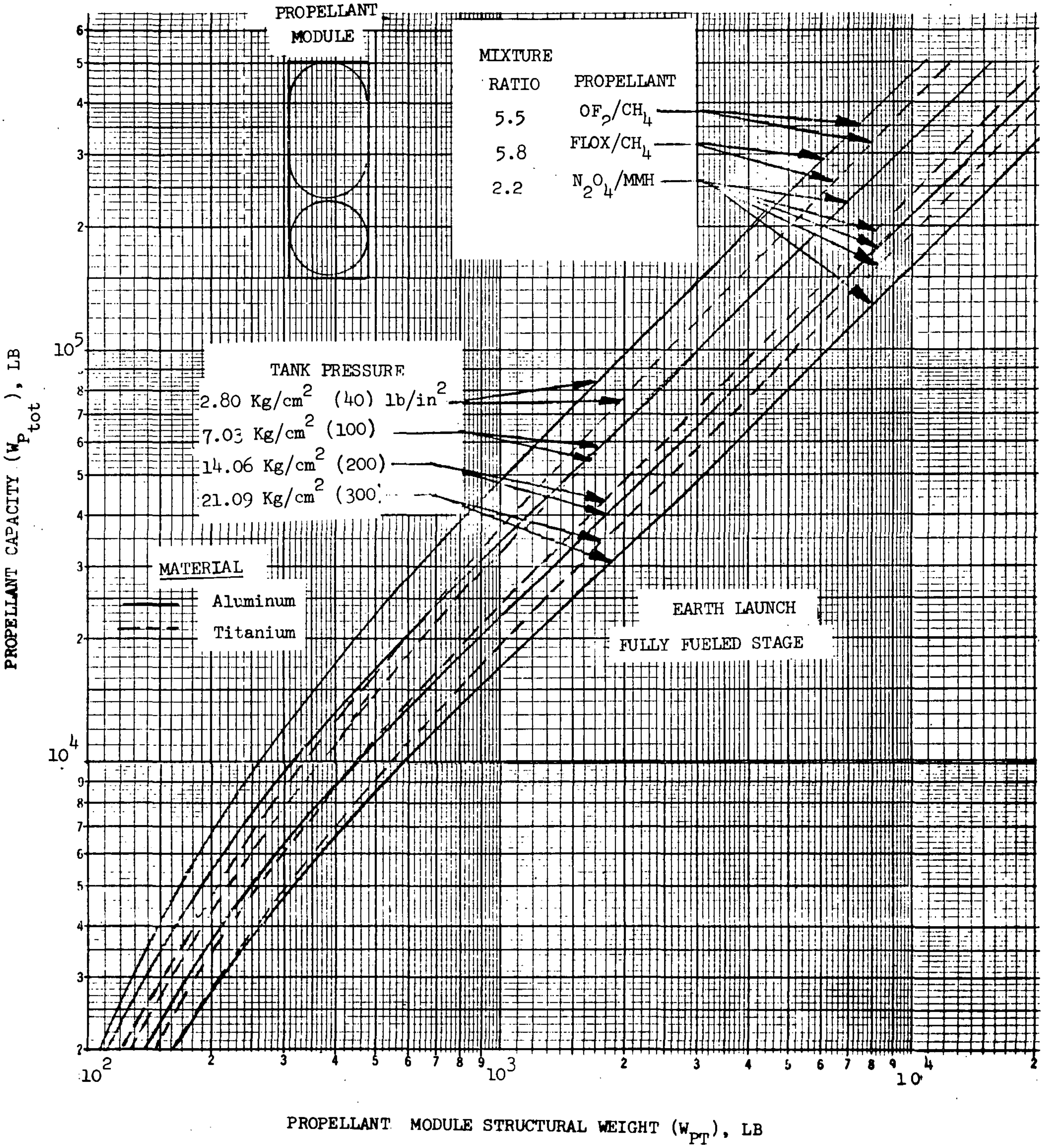


Figure 3. Propellant Module Structural Weight - Storable Fuels

$W_{PR_{ox}}, W_{PR_f}$ = fraction of usable (oxygen, fuel) trapped in plumbing, sumps

$W_{PRV_{ox}}, W_{PRV_f}$ = fraction of usable (oxygen, fuel) provided for contingencies

UF_{ox}, UF_f = tank ullage factor, or fraction of tank volume (based on propellant requirements) added to provide for gaseous oxidizer or fuel

ρ_{ox}, ρ_f = density of (oxidizer, fuel) at estimated tank pressure and propellant boiling-point temperature just prior to ignition.

W_{PR} may be taken equal to 0.5% for small tanks (diameter less than ten feet) or 0.25% for larger tanks.

W_{PRV} will vary between 1% and 5% depending on mission planner's assessment of accuracy of velocity requirements for mission. One method often used is to provide an increase of 0.75% in the velocity increment V and thus set W_{PRV} to zero.

UF_{ox}, UF_f are generally taken as 3% to 5%

The propellant tankage weights were obtained from the SPASM synthesis program. The weight, W_{PT}, include weights for the tank bulkheads and walls, baffles, tank/shell intersection, forward and aft skirts and intertank structure.

Pressurization System Weight (W_{PRESS})

The pressurization system weights are for the pressurant gases, pressurant tankage if any, and the pressurant transmission (plumbing, valves, etc.). One scaling law is used to represent the combination of both pressurization systems for the bi-propellant stages.

Weight for the pressurization systems is

$$W_{PRESS} = K_1 K_2 \frac{W_P}{\rho^*}$$

ρ^* = the propellant combinations bulk density

$$= \frac{(MR+1)\,\rho_{ox}\,\rho_f}{\rho_{ox} + MR\,\rho_f} \quad ; \quad kg/m^3\ (lb/ft^3)$$

where

$$K_2 = 1.0 \quad \text{for steady state continuous pressurization}$$

$$K_2 = \left(\frac{MR}{MR+1}\right)\left(\Delta BO_{ox} + UF_{ox}\right) + \left(\frac{1}{MR+1}\right)\left(\Delta BO_f + UF_f\right)$$

for engine start pressurization only

Table 8. Pressurization System Weight Coefficients

Propellant	Pressurant	Pressure kg/cm^2	(lb/in^2)	K_1 English	K_1 Metric
Cryogen	Helium	2.81	(25)	0.400	6.40
	Nitrogen	2.81	(25)	0.625	10.00
Space	Helium	2.81	(25)	0.335	5.35
Storable	Nitrogen	2.81	(25)	0.525	8.40
Earth	Helium	7.03	(100)	0.775	12.40
Storable	Helium	14.06	(200)	1.525	24.4
Storable	Helium	21.09	(300)	2.275	36.4
Storable	Nitrogen	7.03	(100)	1.225	19.1
Storable	Nitrogen	14.06	(200)	2.425	38.8
Storable	Nitrogen	21.09	(300)	3.625	58.0

Propellant Feed System Weight (W_{PF})

The propellant feed systems for both the oxidizer and fuel tanks are combined into one simplified scaling law, which is given by

$$W_{PF} = 100 + 2.5 \times 10^{-3} W_P$$

4.3 Engine Module Weight

The engine module weight W_{EM} is the sum of the weights of the engine, W_{ENG}, the thrust structure, W_{TS}, and of the shell enclosing the engine, W_t.

$$W_{EM} = W_{ENG} + W_{TS} + W_t$$

Engine Weight (W_{ENG})

Thrust/weight ratios for the different engine types shown in Figure 4 include the thrust chamber assembly and the thrust vector control

$$W_{ENG} = \frac{F\, K_p K_\varepsilon}{(T/W)_{ENG}}$$

K_p and K_ε are modification factors to account for changes in the chamber pressure and expansion ratio, Figure 4 . K_p for pump fed engine systems can be assumed to be equal to 1.0 for the normal range of operating pressure. The weights for high-pressure shuttle-type engines are quoted as a separate curve in Figure 4 . The stage thrust level required can be estimated from the given initial gross weight W_0, or from a first estimate for W_0 when the payload is given. In the latter case, for an initial guess on ν_B,

$$W_0 = W_{PAY} + W_P \left(\frac{1 + \Delta BO}{\nu_B} - \Delta BO \right); \text{ (Payload given)}$$

For departure from an Earth parking orbit, a value of 0.4 to 0.6 for the ratio of engine thrust/gross weight is generally close to that required for maximum performance. For capture at or escape from target planets, the optimum value of F/W_0 varies with the mass of the planet and with the radius and eccentricity of the capture or departure orbit. For orbits no smaller than 2 planet radii, the ratio of F/W_0 (in Earth g's) that provides essentially maximum performance at target planets lies between about 0.2 and 0.4.

The engine thrust level, F, required can be obtained from

$$F = \left(\frac{F}{W_0} \right) \frac{W_0}{N_E}$$

where N_E = the number of engines per stage

Pressure-fed engine weights quoted are for an ablative/radiation-cooled nozzle. Engines with an all ablative nozzle would increase the engine weight by 25%. Propellants of the halogen-family used in the pressure-fed engine system require an additional 20% weight for the change of nozzle material.

Thrust Structure Weight (W_{TS})

Weight of the thrust structure W_{TS} is a function of the total thrust level and the number of engines.

$$W_{TS} = 3.6 \times 10^{-3}\, F\, (N_E)^{0.3}$$

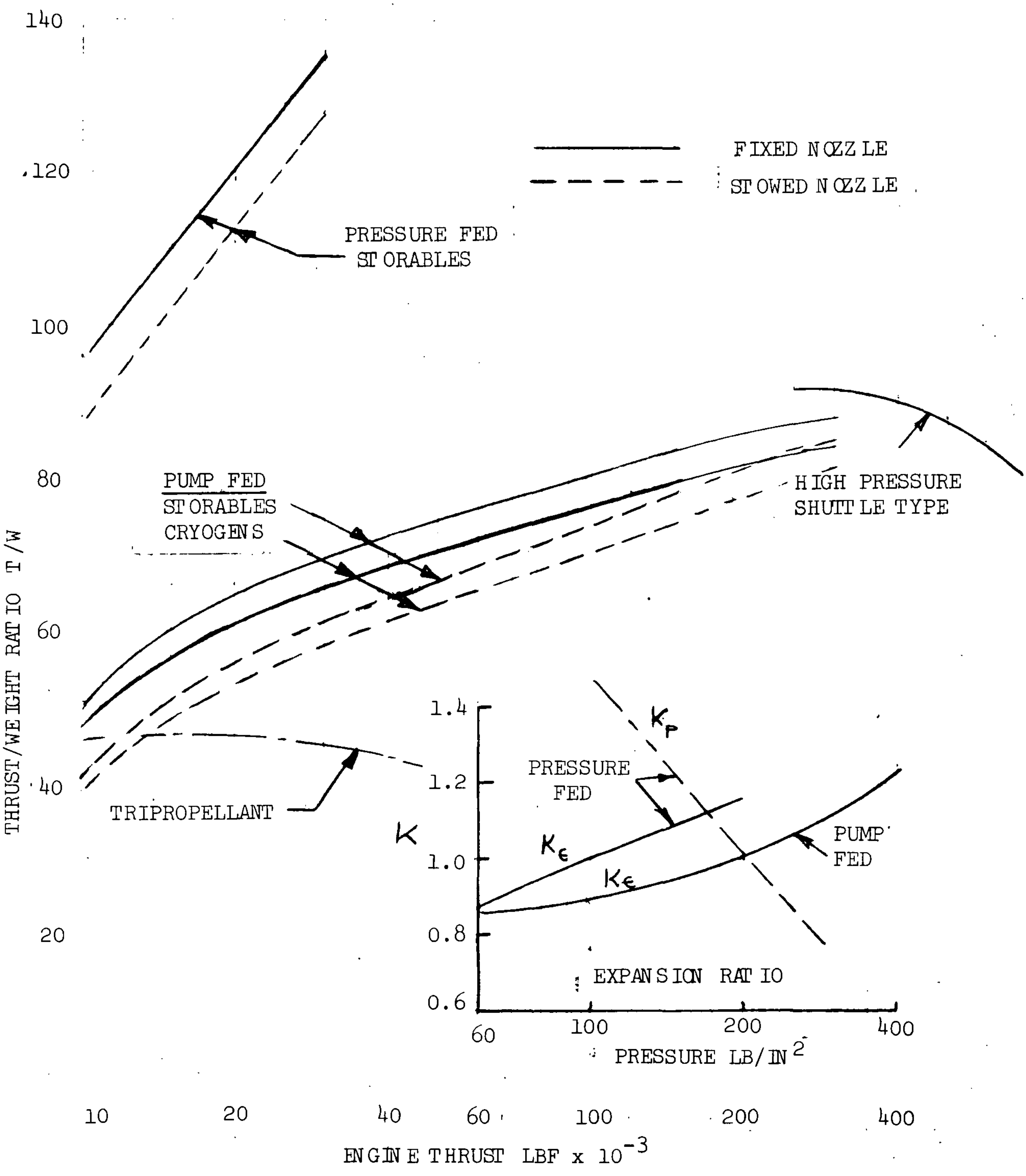


Figure 4 . Engine Thrust/Weight for Different Engine Classes

Outer-Shell Weight (W_t)

The weight of the outer shell W_t of the engine module is a function of the stage diameter and engine length:

$$W_t = K_1 \, W_p{*}^{K_2} \left(\frac{D}{2}\right)^{K_3} L_{ENG} \quad (\text{kg}, \text{lb})$$

where

K_1, K_2, K_3 are given in Table 9

and the stage diameter D is the larger of the values obtained from Figure 5 with a selected L/D and volumes obtained from one or another of

$$V_{OX} = \frac{W_{P_{tot,ox}}}{\rho_{ox}}$$

$$V_F = \frac{W_{P_{tot,ox}}}{\rho_{ox}}$$

Separate tanks for oxidizer, fuel

or from

$$V = \frac{W_{P_{tot}}}{\rho *}$$

, Single tank with common bulkhead

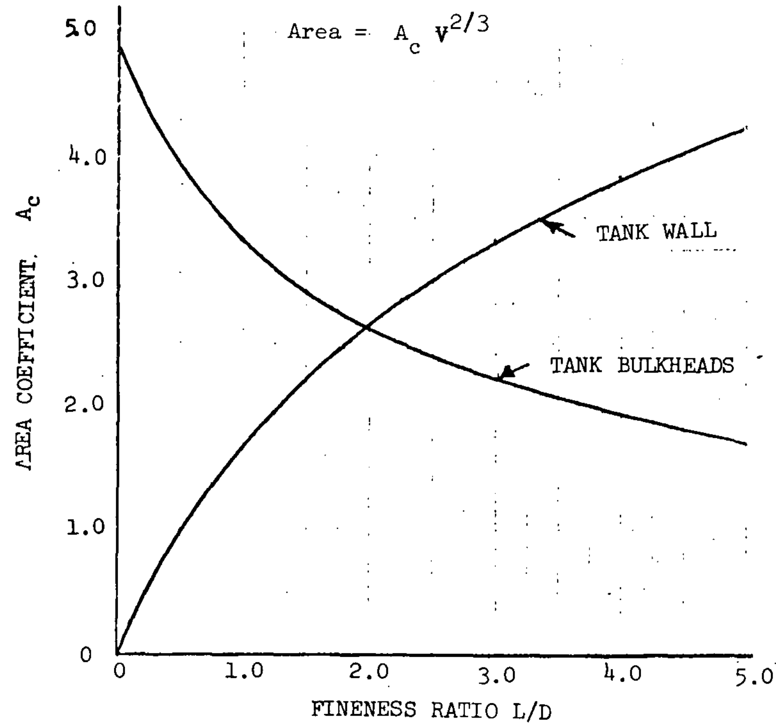
Table 9. Outer Shell Weight Coefficients

Material	K_1	K_2	K_3	
Aluminum	1.76×10^{-4} 5.92×10^{-4}	0.47 0.47	0.85 0.85	Metric English
Titanium	6.7×10^{-5} 2.45×10^{-4}	0.36 0.36	0.85 0.85	Metric English

The length of the engine module is obtained from Figure 6 by selecting the thrust level, chamber pressure and expansion ratio for the engine.

4.4 Environmental Module Weight

The inert weight of the environmental shielding module, W_{EMP}, consists of the total weight penalty for the meteoroid shielding, $W_{MP_{tot}}$, and the stage insulation weight, $W_{INS_{tot}}$.



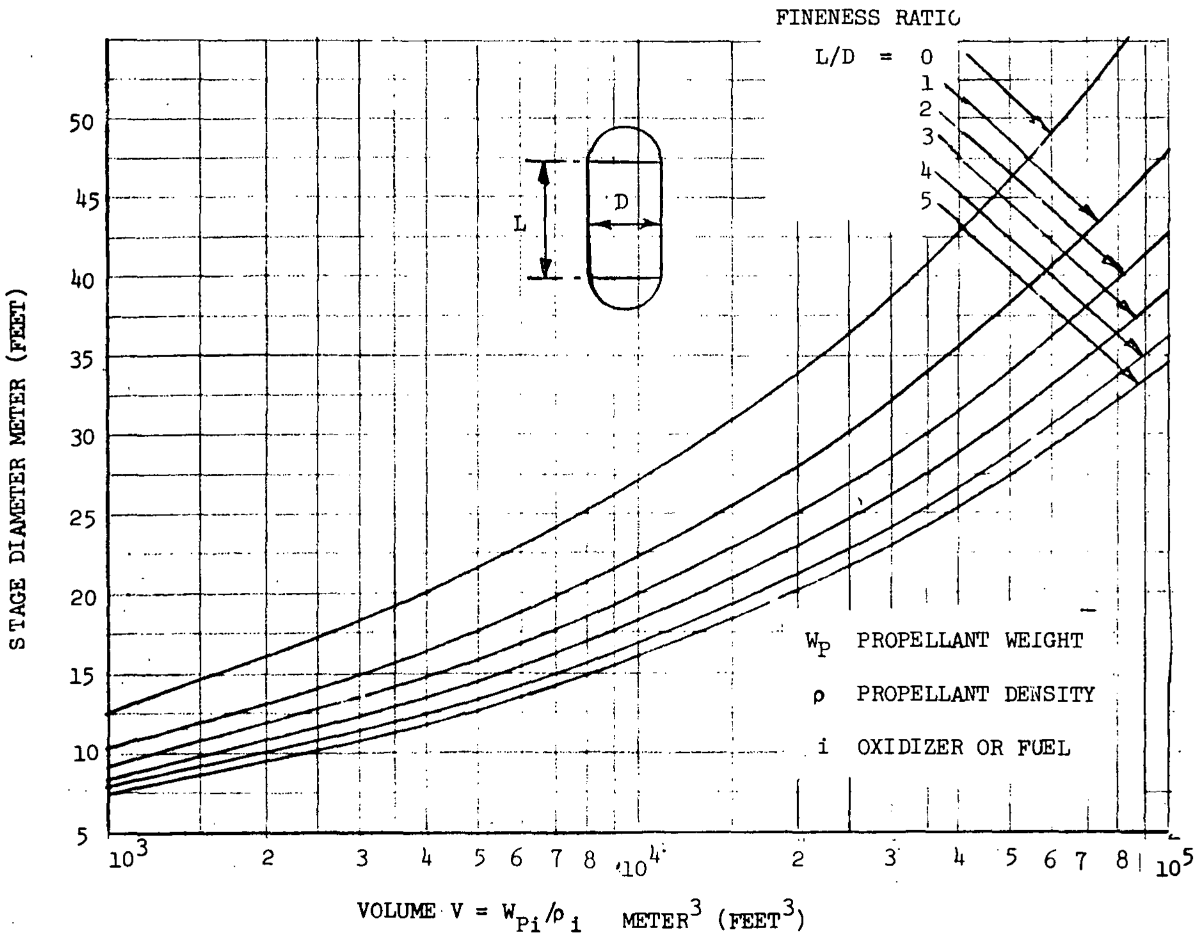


Figure 5. Tank Diameter Versus Volume and Fineness Ratio

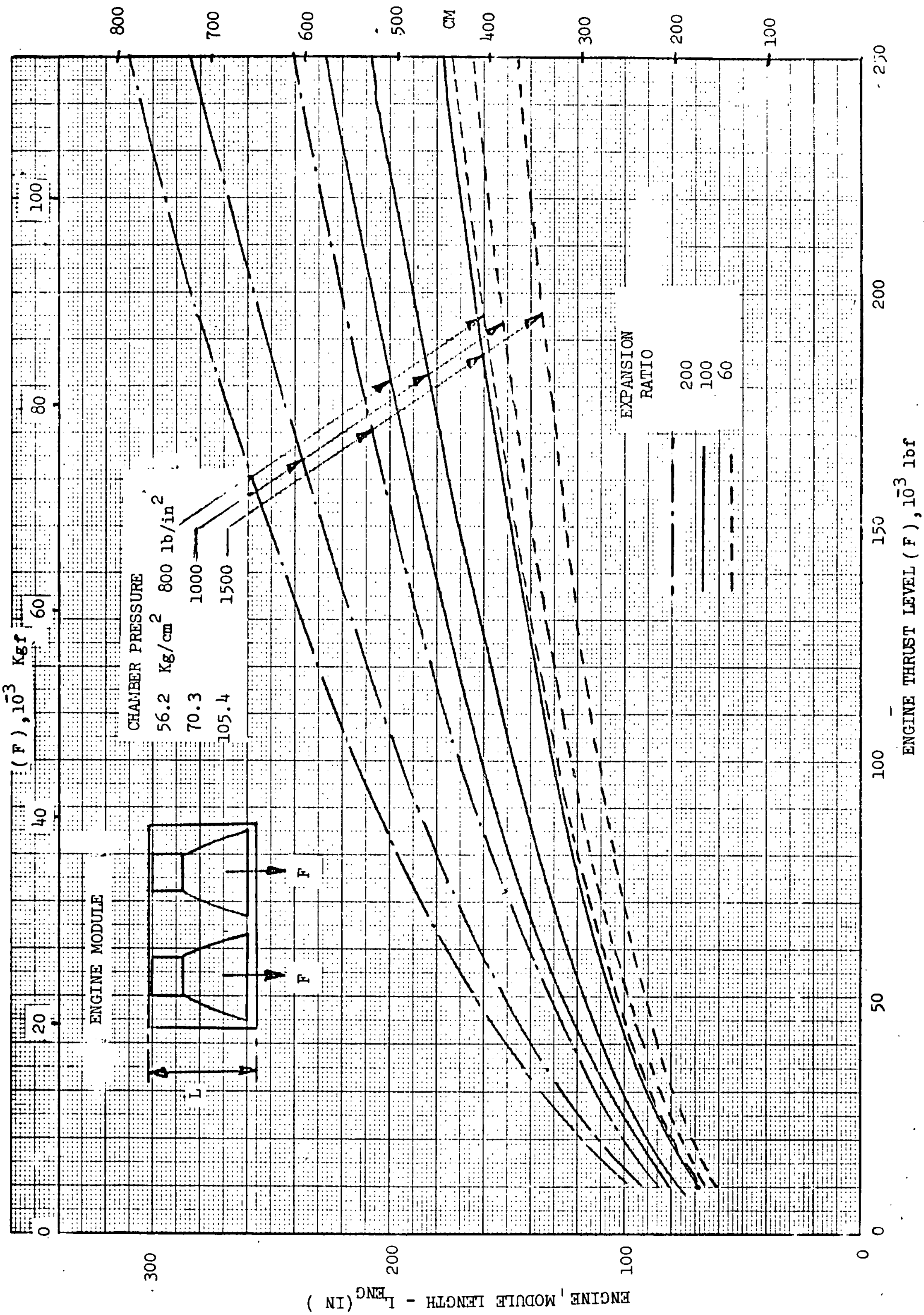


Figure 6. Engine Module Length

$$W_{EMP} = W_{MP_{tot}} + W_{INS_{tot}}$$

Meteoroid Shielding Weight ($W_{MP_{tot}}$)

It is necessary to use a series of figures and charts to evaluate the meteoroid environment, particle diameter and finally the shielding weight requirements. The accuracy of estimation will depend upon graphical interpolation and should be within one or two percent error for the unit shielding weight estimation.

The results obtained from the synthesis program SPASM for the meteoroid flux integration have clearly indicated that there is a strong dependency upon the mission duration and the mission profile. Figures 7 through 10 show that the flux integrals and flux-velocity integrals are non-linear with mission duration and trajectory semi-major axis.

A simplified worksheet, Table 10 is provided and the look-up procedure is capable of handling multi-mission legs with varying flux density, asteroidal and cometary particles, single sheet, single and dual bumper design concepts, different materials and optimizing the overall penetration requirements.

The procedure is identified for the weight estimation together with the data source.

The mission leg parameters are identified as the solar distance of the arrival planet, planet stop-over time, periapse, r_1, and apoapse, r_2, around the planet. The average modification factor is

$$G\eta_{AV} = 8.3 \frac{G\eta_2\left(\frac{r_2}{r_p}\right) - G\eta_1\left(\frac{r_1}{r_p}\right)}{\left(\frac{r_2}{r_p} - \frac{r_1}{r_p}\right)}$$

where r_p = the planet radius

$G\eta_1$, $G\eta_2$ are given in Figure 11

The undisturbed flux at planet distances, Table 11 is corrected by the flux modification factor $G\eta_{AV}$ to obtain FLX_A and FLX_C.

The velocity integral is evaluated by

$$V \times FLX_A = \overline{V} \times FLX_A$$

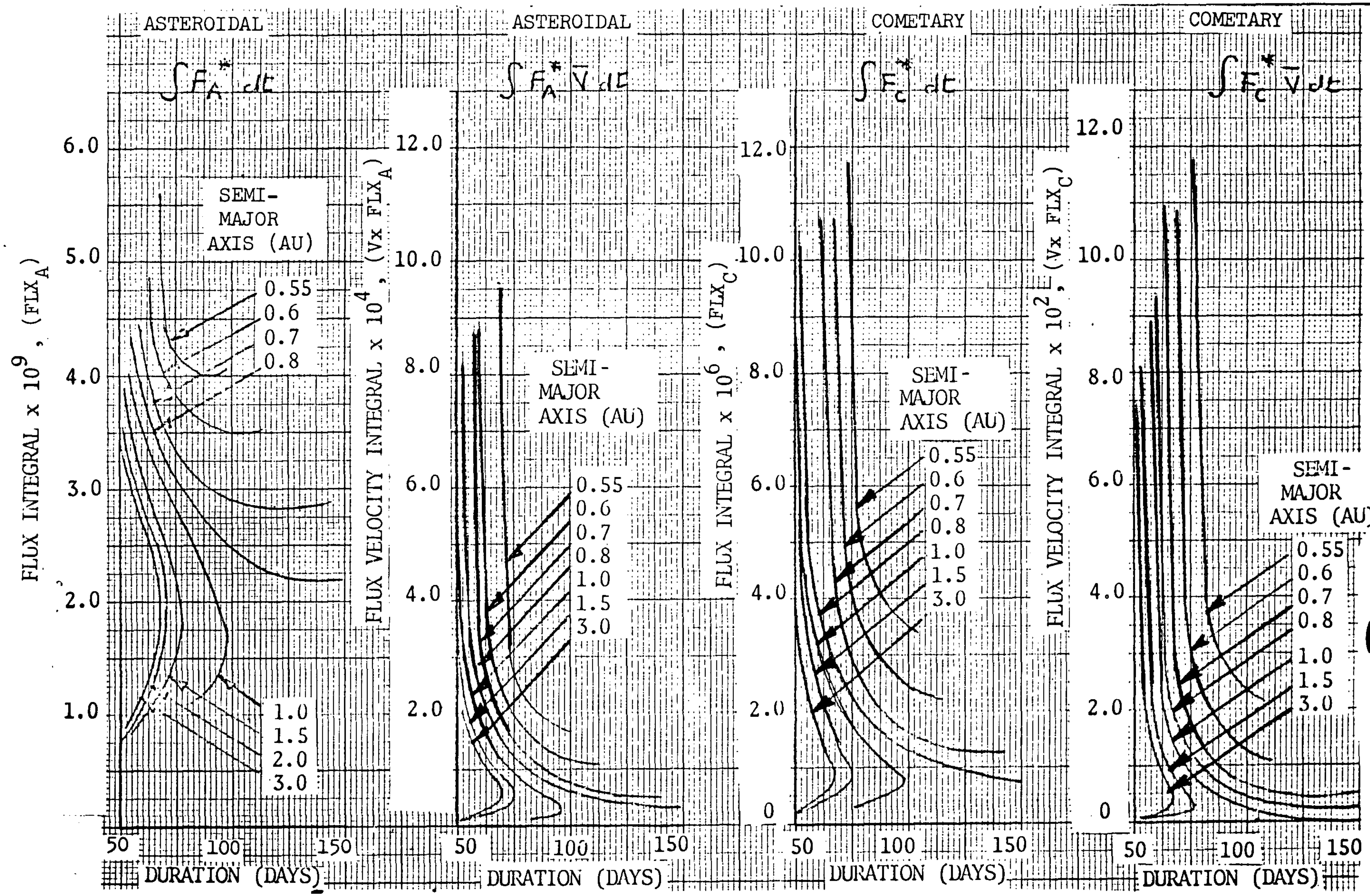


Figure 7. Flux and Flux Velocity Integral - Earth/Venus Mission

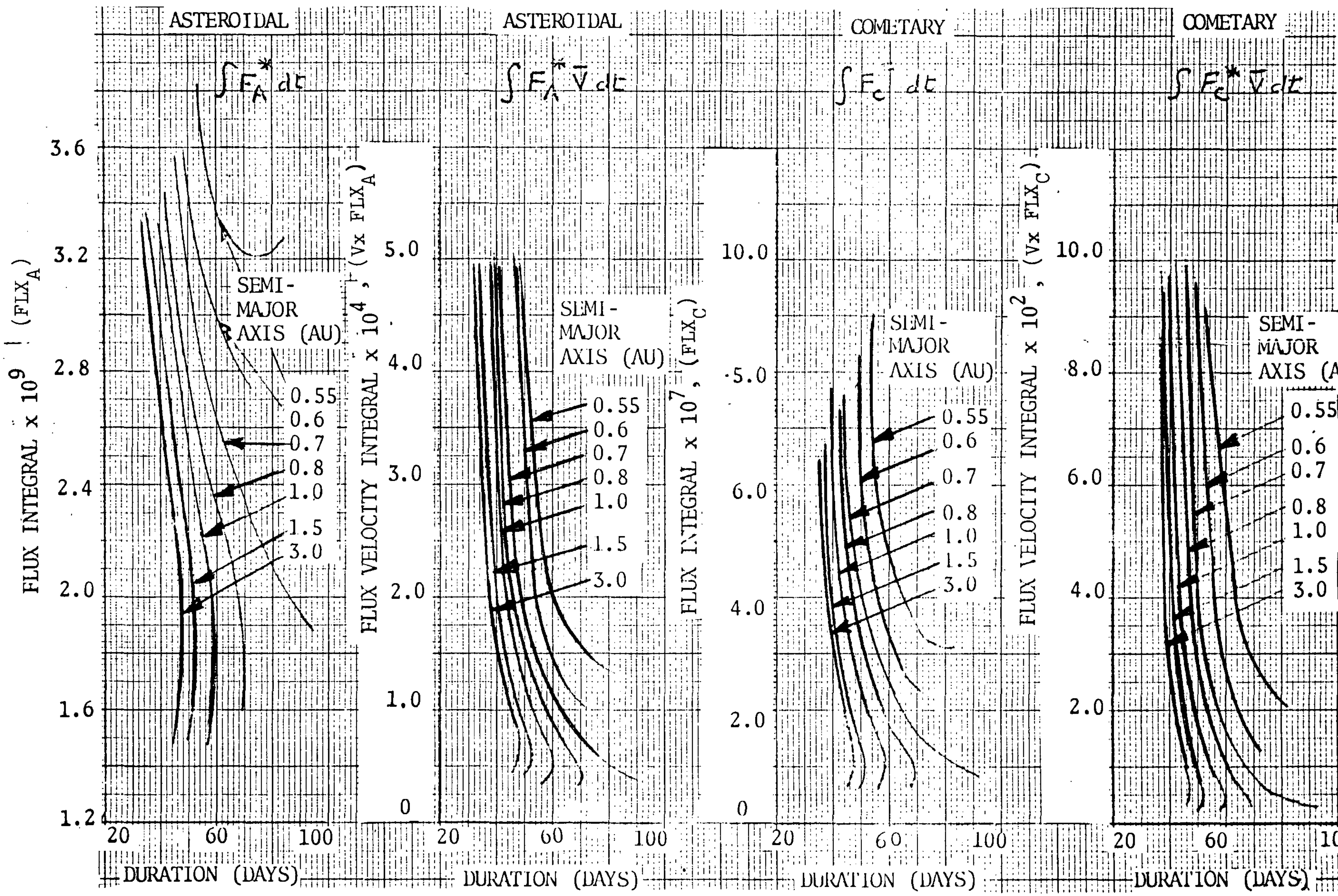


Figure 8. Flux and Flux Velocity Integral - Earth/Mercury Mission

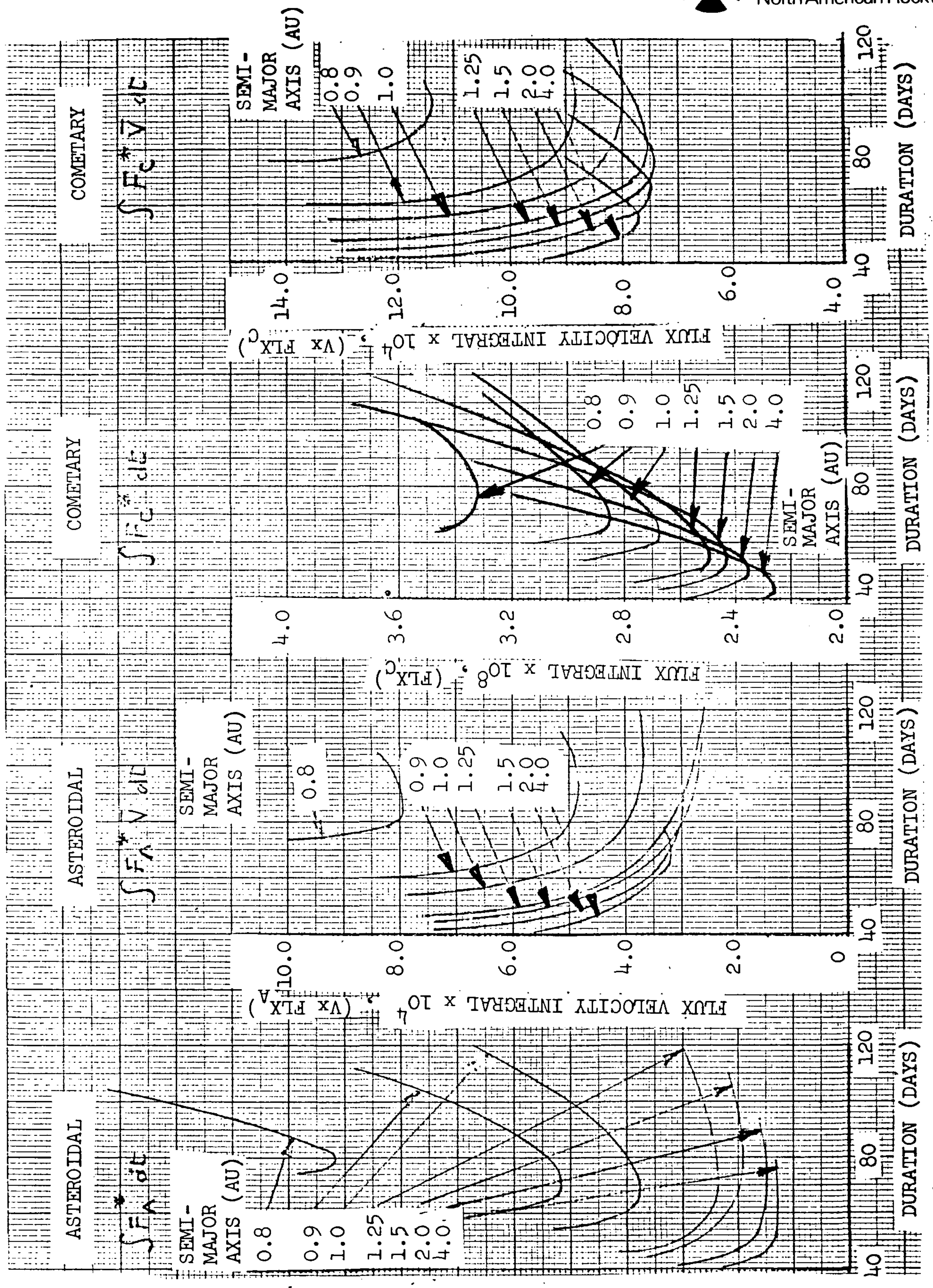


Figure 9. Flux and Flux Velocity Integral - Venus/Mars Mission

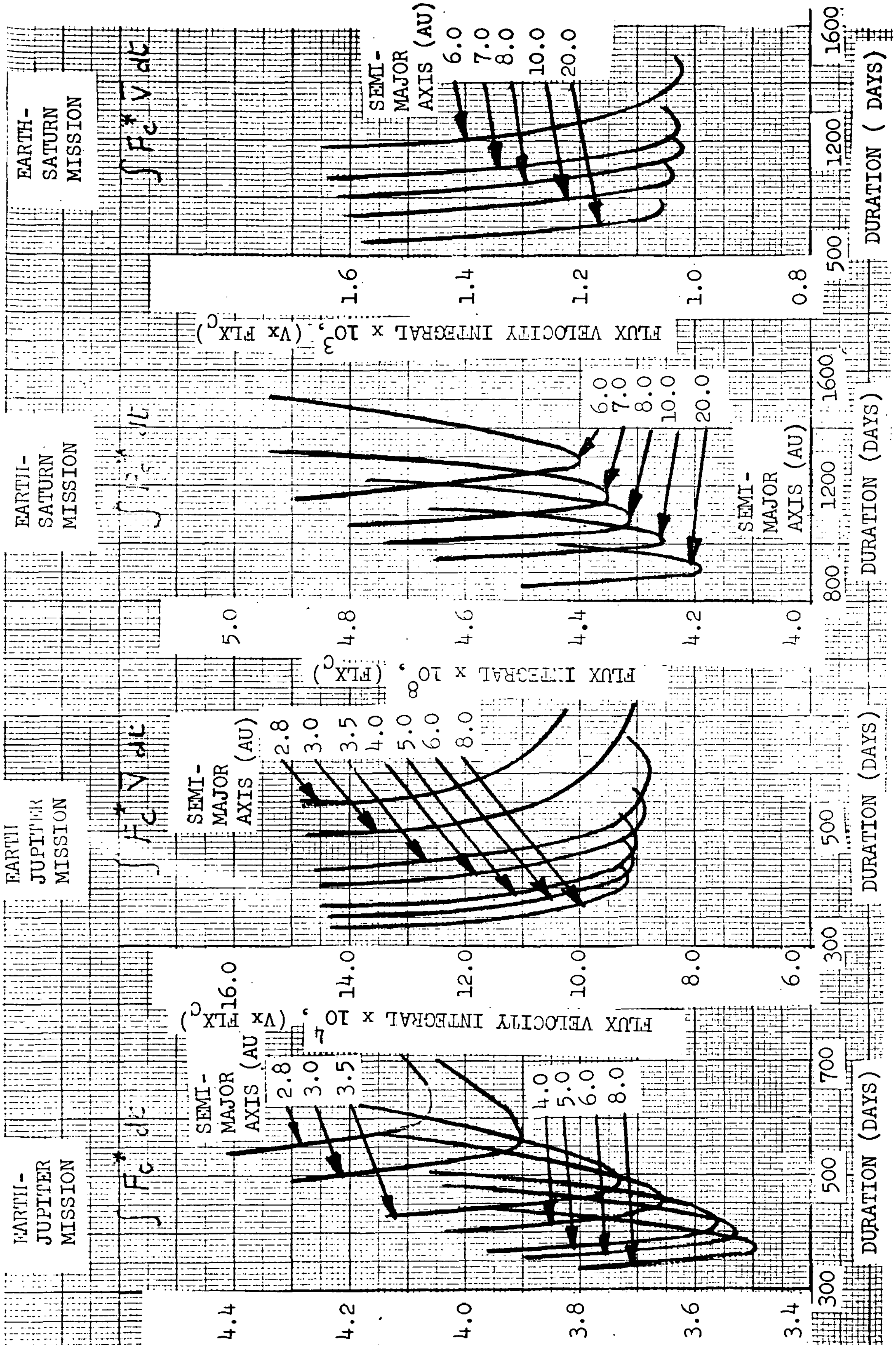


Figure 10. Cometary Flux and Flux Velocity Integral - Earth/Jupiter and Earth/Saturn Missions

Table 10 Meteoroid Shielding Requirements

MISSION FLUX INTEGRATION

	MISSION LEG STAGE IS EXPOSED				
	1	2	3	4	Total
Departure Planet Arrival Planet Solar Distance AU Planet Orbit Radii Stop-Over Time (hrs) Modification Factor G_η					
Sporadic $V \times FLX_A$ Asteroidal FLX_A					
Sporadic $V \times FLX_C$ Cometary FLX_C					

SHIELDING WEIGHT

	SPORADIC ASTEROIDAL	SPORADIC COMETARY
Average Velocity, m/sec Stage Surface Area, m^2 Po of No Penetration Particle Diameter , cm Diameter Factor α Bumper Wt , W_B; Kg/cm^2 Rear Sheet Factor W_m/d^α Rear Sheet Wt , W_m; kg/cm^2		

$P_o = P_{o_a} \times P_{o_c} =$

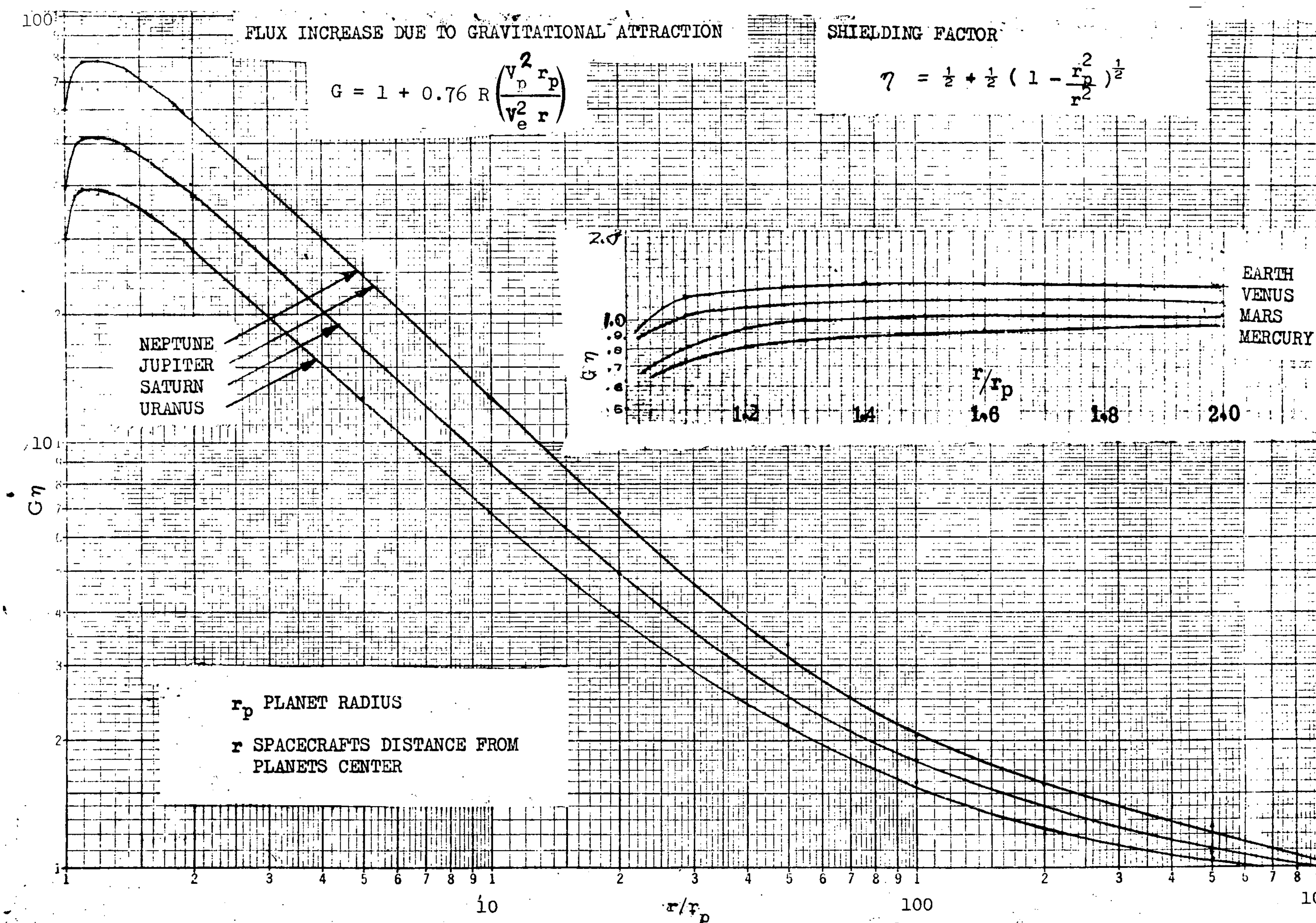


Figure 11 . Meteroid Modification Factor Due to Planets

where $\overline{V}$ is given in Table 11.

Table 11. Meteoroid Fluxes at Planet Distances

PLANET	COMETARY FLUX		ASTEROIDAL FLUX	
	F_{PC}* $\frac{\text{Particles}}{m^2 \text{ Year}}$	$\overline{V}$ m/sec	F_{PA}* $\frac{\text{Particles}}{m^2 \text{ Year}}$	$\overline{V}$ m/sec
Mercury	6.822×10^{-7}	31000	-	14900
Venus	1.955×10^{-7}	22700	1.399×10^{-10}	10900
Earth	1.021×10^{-7}	19300	3.761×10^{-9}	9300
Mars	4.399×10^{-8}	15600	6.365×10^{-8}	7500
Jupiter	3.774×10^{-9}	8450	1.0105×10^{-10}	2580
Saturn	1.123×10^{-9}	6240	-	1850
Uranus	2.777×10^{-10}	4400	-	1300
Neptune	1.131×10^{-10}	3500	-	1040
Pluto	6.569×10^{-11}	3070	-	910

The transplanetary mission segment flux and flux-velocity integrals for a particular planet pair are obtained from Figures 7 through 10.

The procedure is repeated for each mission leg and the total mission flux is the sum of all the flux integrals and the weighted particle velocity V_P is

$$V_{Pa,c} = \frac{\Sigma \; V \times FLX_{a,c}}{\Sigma \; FLX_{a,c}}$$

A representative exposed surface area for each stage is considered based on the initial estimate of the propellant volume.

$$A = \left(A_{cox_{BULK}} + A_{cox_{WALL}}\right) V_{ox}^{2/3} + \left(A_{cf_{BULK}} + A_{cf_{WALL}}\right) V_f^{2/3} + \pi \, D \, L_{ENG} \quad ; \; M^2$$

Area coefficients for both tanks are obtained from the engine module section using Figure 5 .

A probability of no penetration for the cometary flux (P_{oc}) is selected where $P_o < P_{oc} < 1$ and the meteoroid particle for design purposes is obtained from

$$d_{Pa,c} = \left[\frac{6}{\pi \rho_{a,c}} \left(\frac{1-P_{oa,c}}{A \sum \int F^*_{a,c}\, dt} \right)^{1/K_{2a,c}} \right]^{1/3}$$

where K_2 = -0.84 (asteroidal); -1·213 (cometary)

ρ = 3.5 (asteroidal); 0.5 (cometary)

For the single and dual bumper design concepts, the unit weight of the outer bumper, W_B, is given in Figure 12 and depends on the particle diameter and the bumper material.

The rear sheet weight requirements can be determined with the aid of Figure 13 and the meteoroid impact velocity V_p. An exponent α associated with the particle diameter is given in Table 12 as a function of the material and the shielding concept. The unit weight for the rear sheet is

$$W_{M2} = \left(\frac{W_{M2}}{d^{\alpha}} \right) d^{\alpha}$$

Table 12. Meteoroid Particle Diameter Exponent, α

Meteoroid	Material	Single Sheet	Single Bumper	Dual Bumper
Cometary	Aluminum	1.0535	1.12	1.15
	Titanium	1.0535	1.11	1.09
	Glass Epoxy	-	1.09	-
Asteroidal	Aluminum	1.0535	1.11	1.04
	Titanium	1.0535	1.12	1.12
	Glass Epoxy	-	1.09	-

Actual meteoroid shielding unit weight penalty (W_{mp}) considers the shell material, unpressurized/pressurized and insulation shielding allowances

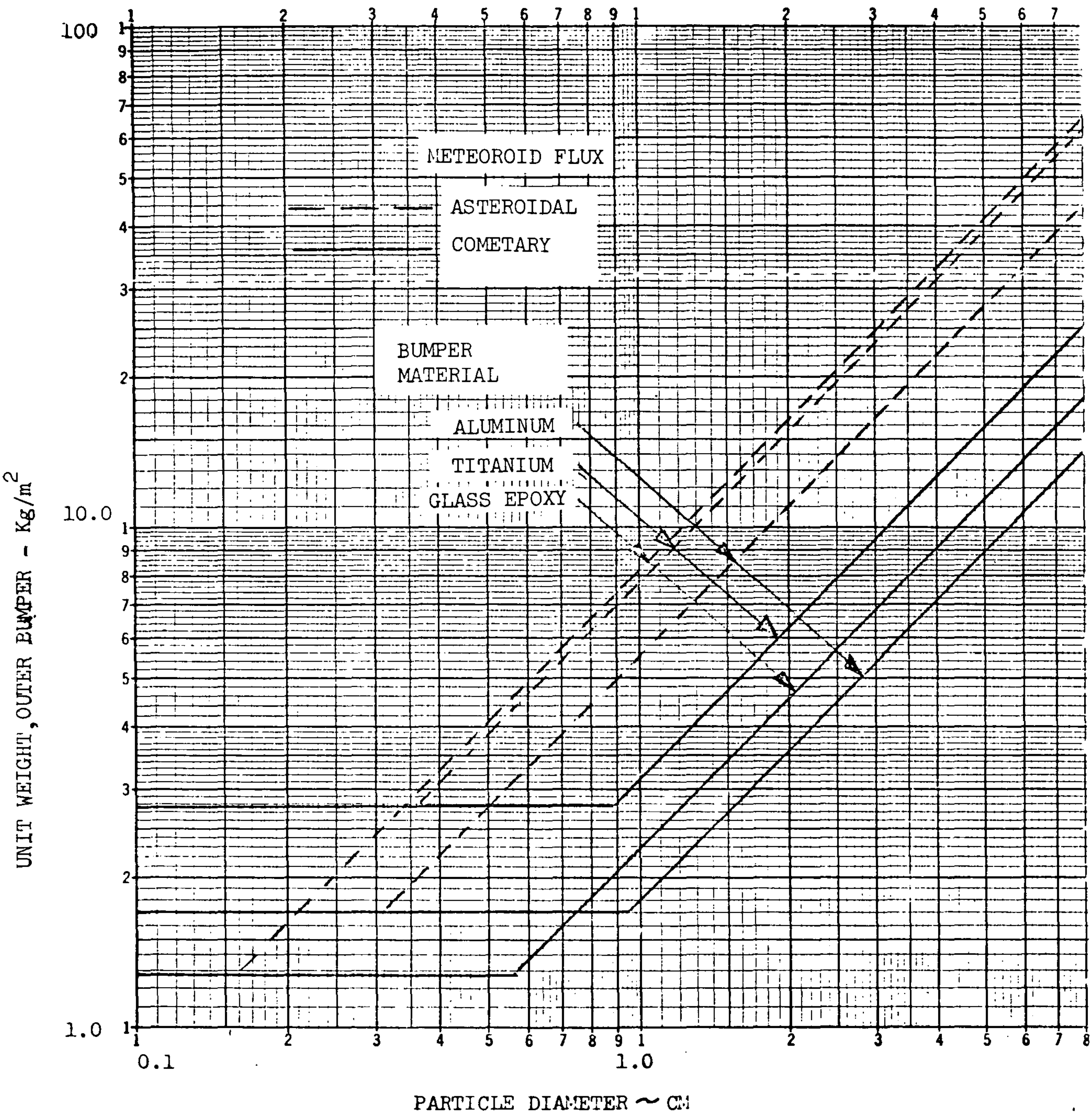


Figure 12. Meteoroid Bumper Unit Weight

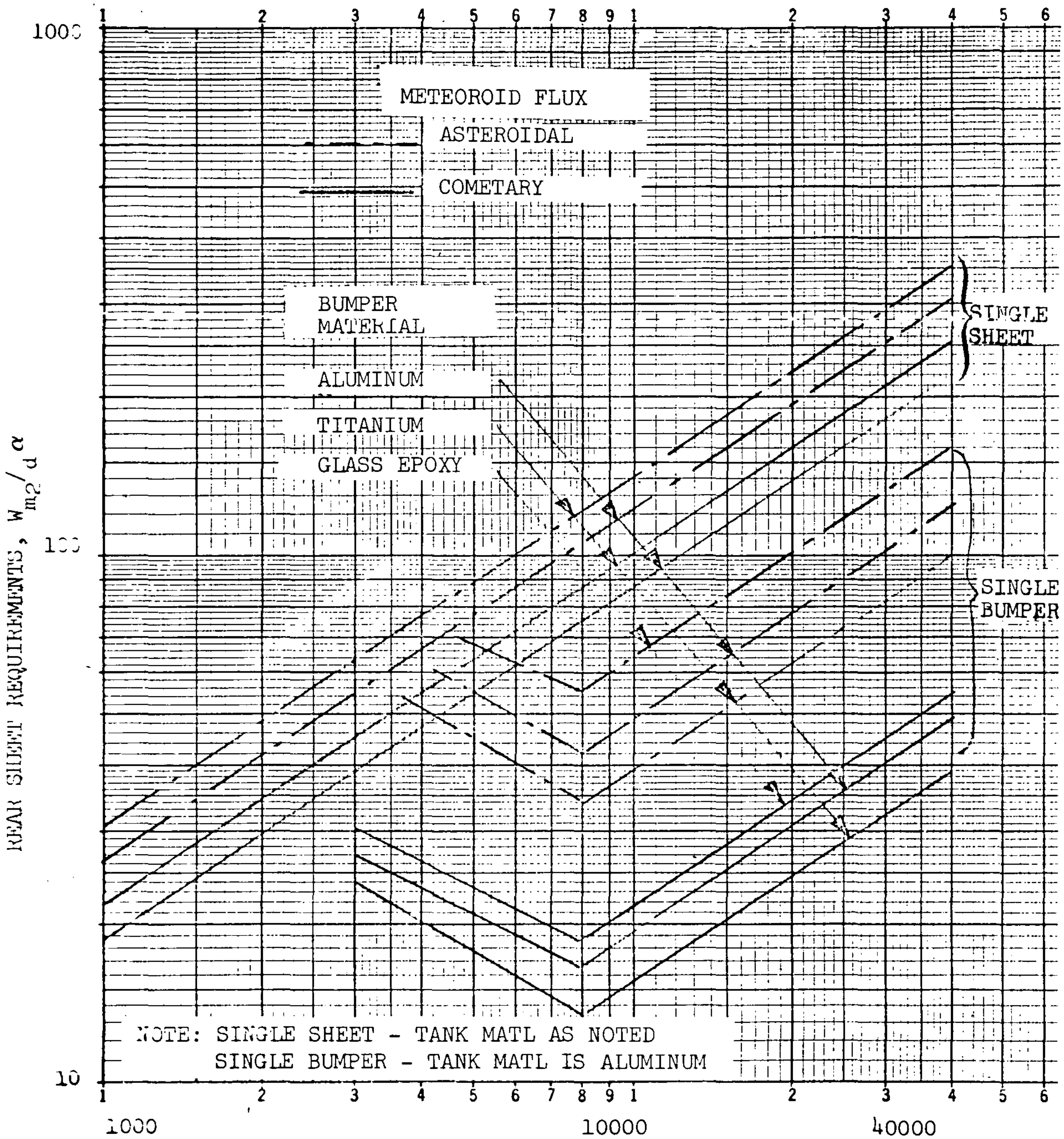


Figure 13. Meteoroid Shielding Rear Sheet Requirements

$$W_{mp} = W_B + \frac{K_1 \, K_2 \, W_{m2}}{\text{EXP } (14.9 \ \rho_{ins} \ \tau_{ins}/d^*)} - W_S$$

where K_1 = 1.0 for pressure tanks

= 0.445 for unpressurized shells

K_2 = 1.0 aluminum

= 1.15 titanium

= 0.83 Glass Epoxy

ρ_{ins} = insulation density (gm/cm^3)

τ_{ins} = insulation thickness (cm)

d^* = max (d_p, 1.0) (cm)

W_S = existing unit weight for structural integrity

The procedure is repeated for the asteroidal flux and the shielding penalty is taken as the maximum W_{mp} due to the cometary or asteroid fluxes. The stage overall mission penetration probability P_o is

$$P_o = P_{oa} \times P_{oc}$$

The total meteoroid shielding weight penalty, $W_{MP_{tot}}$ is given by

$$W_{MP_{tot}} = A_{unp} \, W_{mp_{unp}} + A_p \, W_{mp_p}$$

where subscript unp is for the unpressurized structural elements (skirts, intertank, interstage) and the subscript p refers to the pressurized structural elements (tank walls).

Stage Insulation Weight ($W_{INS_{tot}}$)

A simplified approach for predicting thermal requirements will use simple models and several charts depicting the integrated thermal properties of various insulation/propellant combinations. The following procedure will provide the insulation weight estimates.

Planet stop-over mission legs subject the stage to planet albedo and emitted radiation, B_f,

$$B_f = \left(1 - \frac{\sqrt{h(r_p+h)}}{(r_p + h)}\right)\left(B + 0.25\ (1-B)\right)$$

where R_p = solar distance of planet, AU

B = albedo of planet

r_p = radius of planet

h = average altitude of spacecraft orbit

The equilibrium temperature T_H of the outer surface during a planet stop-over is

$$T_H = K_1 \left[\left(\frac{\alpha_s}{\varepsilon} \right) \left(\frac{1+B_f}{R_P^2} \right) \right]^{1/4}$$

where K_1 = 281 (metric); 505 (English)

α_s/ε = absorptivity/emissivity ratio (0.20)

For the transplanetary mission segments the equilibrium temperature is

$$T_H = K_1 \left[\left(\frac{\alpha_s}{\varepsilon} \right) \left(\frac{1}{R^2} \right) \right]^{1/4}$$

where R = solar distance of the spacecraft, AU

The normalized unit flux,

$$H_n = \int_{T_C}^{T_H} K dT$$

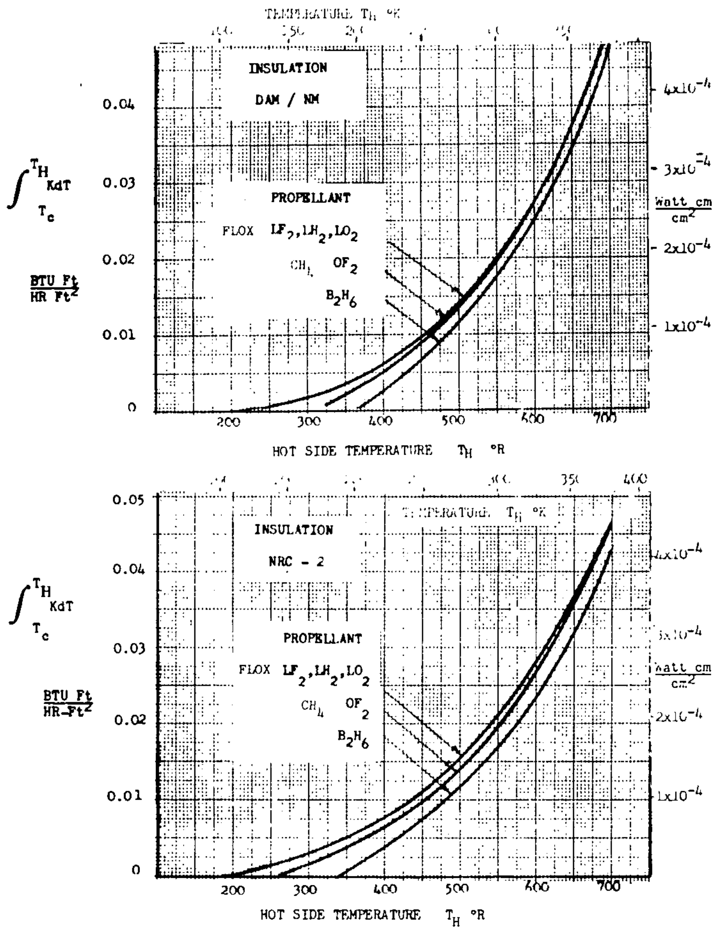
is obtained for each mission leg from Figure 14 for the specific propellant and the type of thermal insulation used for the tank protection. The total flux integral K is

$$K = \sum H_n \times \text{Time} \times (1+C)$$

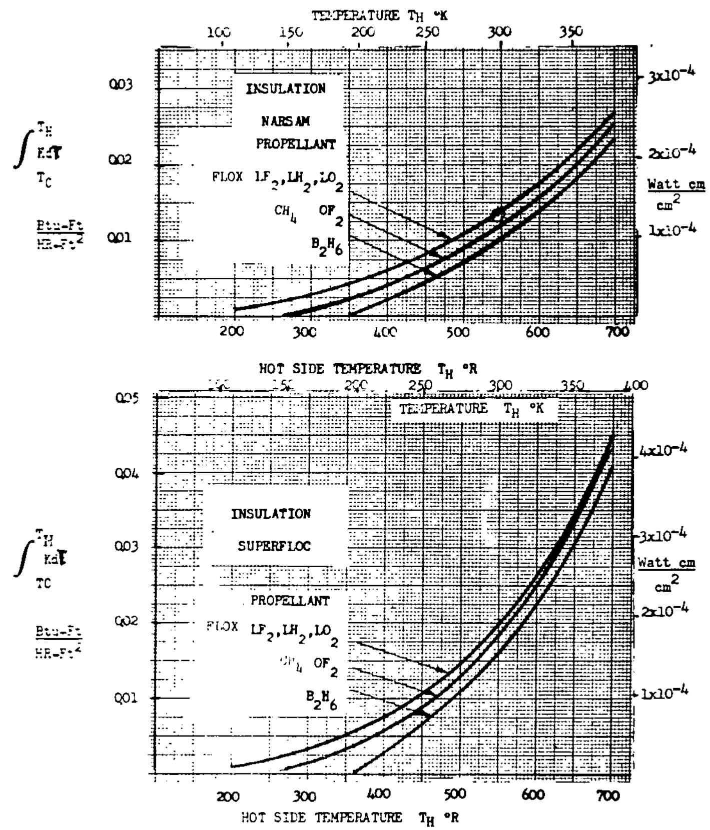
where C = correction factor to account for heat leaks through support structure 0.5 (aluminum); 0.4 (titanium); 0.25 (structure with heat blocks)

A thickness-heating parameter (d/k) is evaluated from the thermal insulation sizing nomograph Figure 15 for a fixed percent propellant boil-off. The required insulation thickness is obtained from

$$d_{INS} = (d/k)\ K$$



Heat Flux for DAM/NM & NRC-2 Insulation



Unit Heat Flux for NARSAM & Superfloc Insulation

Figure 14.. Unit Heat Flux With High Performance Insulation

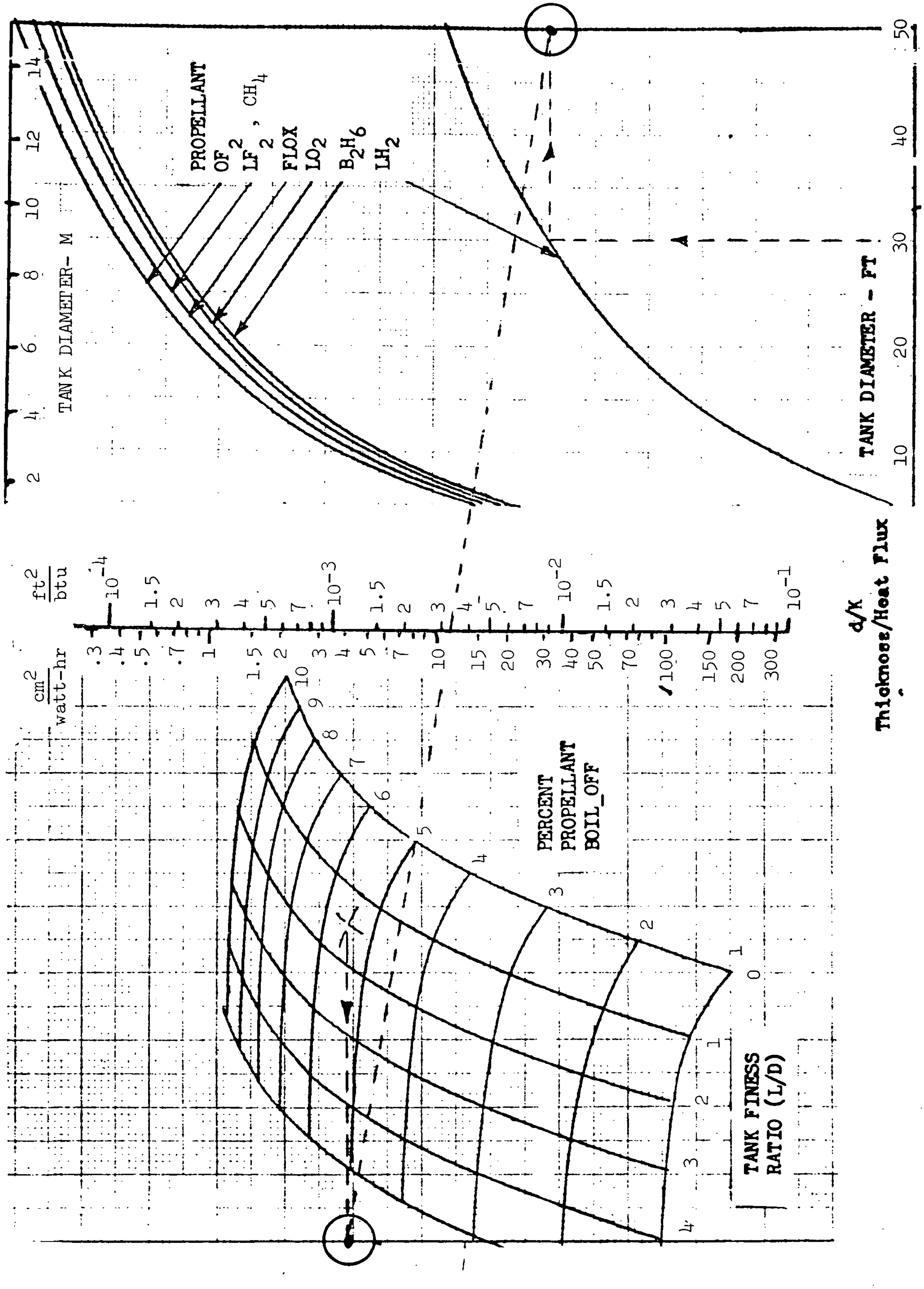


Figure 15 . Thermal Insulation Sizing Nomograph

The stage insulation weight is given as

$$W_{INS_{tot}} = \left(\rho_{INS}\, d_{INS} + W_{INS}\right) A_T$$

where

A_T = thermal protected area including tank walls (A_P) and bulkheads (A_B) (if exposed) and unpressurized shells (A_{unp})

A_P = $A_{COX_{WALL}} \; V_{OX}^{2/3} + A_{cf_{WALL}} \; V_f^{2/3}$

A_B = $A_{COX_{BULK}} \; V_{OX}^{2/3} + A_{cf_{BULK}} \; V_f^{2/3}$

A_{unp} = $A_B + \pi D L_{ENG}$

W_{INS} = 6×10^{-5} kg/cm^2; 0.12 lb/ft^2 for ground hold

= 2×10^{-4} kg/cm^2; 0.40 lb/ft^2 if insulation is exposed to aerodynamic pressure forces during the Earth boost

= 0.0 for space exposure with an outer meteoroid bumper and orbiting fueling

<u>No Propellant Boil-off - Increased Pressure</u>

The allowable heat input per unit volume of propellant Q/V_L, is evaluated for a particular propellant tank pressure, P, and a specified ullage percent (UF).

$$Q/V_L = K_1 \; \mathrm{EXP} \left(\frac{K_2}{K_3 - P}\right) UF^{1.25}$$

Table 13. Heat Input Coefficients

	K_1				K_2	K_3
Propellant	LH_2	LO_2	LF_2	B_2H_6		
Metric	0.315×10^5	2.93×10^5	3.7×10^5	3.15×10^5	0.945	1.05
English	0.306×10^4	2.84×10^4	3.6×10^4	3.06×10^4	13.5	15.0

The required insulation thickness is given by

$$d_{INS} = \frac{K}{(Q/V)} \quad \frac{A}{V}$$

where A = surface area of tank

K = total heat flux integral

Insulation Optimization

There are several ways to optimize the insulation requirements; each method based on different performance criteria. The usual method is to minimize the weight of the propellant boiled-off, insulation and additional tank structure and neglect the effect on the stage performance. A second approach is to minimize the vehicles initial weight for a specified mission performance requirement for the vehicle. The optimized insulation thicknesses for a two-stage vehicle (single burn) using the second method are

$$d_{1_{INS}} = \frac{f_1 K_1}{L_1} + \frac{1}{L_1}\sqrt{\frac{L_1 K_1}{\rho_{ins_1}}\left(G_1 + \frac{1}{\mu_1}\right) + f_1^2 K_1^2}$$

$$d_{2_{INS}} = \frac{f_2 K_2}{L_2} + \frac{1}{L_2}\sqrt{\frac{L_2 K_2}{\rho_{ins_2}}\left[G_2 + \frac{1}{\mu_1 \mu_2}\left(1 + \frac{K_{21}}{K_2}(\mu_1 - 1)\right)\right] + f_2^2 K_2^2}$$

The weight of propellant boil-off for each of the two stages with the optimum insulation thickness is

$$W_{B_{i_{opt}}} = \frac{A_i \; K_i}{d_{i_{INS}} \quad L_i - f_i \; K_i} \qquad i = 1 \text{ or } 2$$

Results for a single stage vehicle are identical to the first stage of the two-stage vehicle.

4.5 OTHER SYSTEMS MODULE WEIGHT

The weight estimation for the other systems module, W_{SYS}, can be obtained from the equation and empirical coefficients detailed in Section 3.2.5.

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5.0 CONCLUSIONS AND RECOMMENDATIONS

The development of improved scaling laws during this study have illustrated the complex interaction between the mission and vehicle design with regard to estimating the stages inert mass. Simple universal scaling laws would not be adequate to describe all the model variations and combinations required for this study. It is only when the propulsion stage is considered as a series of major components that their weight relationships can be correlated to the important design parameters. Scaling laws for several components are derived solely from empirical data of existing hardware system. Wherever possible, parametric data was developed from models using actual detail design analysis and environment evaluation. The parametric data were subsequently reduced to provide the required scaling laws.

Major emphasis has been placed on investigating the design parameters affecting the weight of the heavier components which contribute to the major portion of the stage's inert mass. Therefore, comprehensive investigation was conducted in four areas -

1) Engine weight data
2) Structural synthesis
3) Meteoroid protection
4) Thermal insulation

It was felt that realistic scaling laws in these four areas which recognized differences in vehicle design and types of mission would provide the best means of estimating the overall stage inert mass.

In order to adequately define the component weights, it is required to consider several design and mission parameters. Design parameters included in the scaling laws are categorized into five different areas, each area containing several specific sub-areas or parameters. A list of the parameters used for this study is shown in Table 16. It is required to have scaling laws which represent the component throughout a large range of vehicle sizes. Most of the scaling laws are composed of several design parameters to adequately describe the components weight. It was found that a single scaling law could not be applied for some components for their entire range of design parameters. Therefore piecewise scaling laws are provided and their particular range of application identified.

Table 16. Study Design Parameters

	Design Parameters
1. Mission Objectives	1) Velocity requirements 2) Trajectory profile 3) Transplanetary & planet stop over duration 4) Meteoroid flux 5) Thermal flux
2. Design Characteristics	1) Number of stages 2) Number of engines 3) Tankage arrangement 4) Shielding concepts
3. Propulsion/ Propellant Characteristics	1) Type of engines a) Pressure or pump-fed b) Thrust level c) Chamber pressure d) Expansion ratio, fixed or stowed nozzle 2) Propellant combination a) Cryogenic, storables b) Mixture ratio
4. Structural Characteristics	1) Construction concept a) Ring stiffened/waffle b) Single/multiple bumper 2) Material a) Stress allowables b) Modulus and density
5. Design Criteria	1) Temperature 2) Design loading a) Pressure b) Axial loading c) Bending moments 3) Factors of safety

The scaling laws were developed as exponential equations wherever possible. An equation of this form is monotonic and can be sculptured to fit most arrays of weight data by suitable adjustment of the coefficients and parameter exponents. A weighted multi-linear regression analysis applied to the weight data would minimize the percentage error across the ranges of the design parameters. This approach is superior to that of minimizing the overall errors, which would result in poor weight estimates for the small vehicles and the estimation would only improve for the large vehicles.

It was found that the propellant module weight cannot be represented by a single law which is a function of the stage propellant weight and bulk density. Significant weight variations are present due to design restrictions (stage diameter, fineness ratio), subsystem selection and design criteria, that additional modification factors are required to the basic scaling law. These modification factors are treated as linear and independent factors. Additional effort is required to further identify the interaction between the modification factors and how they vary throughout the design spectrum.

An important weight item is the meteoroid shielding which is strongly dependent on the mission and the shielding concept. While it was relatively easy to provide scaling laws for shielding weight as a function of meteoroid particle diameter and velocity, there was no reasonably accurate method of approximating the mission flux environment. Although the SPASM program quickly integrates the mission flux experienced by the spacecraft, the resulting flux is strongly dependent on the time/solar distance relationship and the relative velocity vector between the spacecraft and the meteoroid flux. Results shown in Section 4.4 clearly indicated the unusual patterns for the meteoroid flux integral between several planet pairs. Therefore, for the manual method using the simplified scaling laws, a graphic data presentation has been supplied rather than a simple equation for the flux evaluation. This will not detract from the weight estimation and structural sizing. The flux integral can be evaluated initially for the particular mission using the supplied figures and the result will provide an estimate for the particle diameter and velocity which can then be applied to the weight scaling law for the meteoroid shielding.

A similar condition exists with the thermal insulation where there are too many parameters, mission variations and material properties to relate to a single scaling law. Therefore, with the manual method using the simplified scaling laws, an initial evaluation has to be performed with the help of several figures and a design nomograph to estimate the unit weight of insulation and percentage boil-off (if any) for each particular mission.

The SPASM program developed for this study provides an excellent means of providing weight/sizing data for vehicles subjected to most types of missions and environments. This data can be used to perform trade-off studies to determine the relative performance effectiveness of competing vehicle designs.

It is recommended that the techniques, scaling laws and computer synthesis program developed for the study be applied to NASA on-going and future studies dealing with conceptual design and evaluation of space propulsion systems. The scaling laws provided are representative of the study design models. The final weight values used for other than parametric studies will require examination and verification by Weight Specialists.

The program models and subroutines have been checked individually. The extensive combinations and permutations of the models can be exercised and used in parametric studies recognizing that some routes may require program modification.

Scaling coefficients are provided for recognizable advanced systems. New technology breakthroughs will require update of many of these weight scaling coefficients and perhaps additional subroutines to the synthesis program.